

**Final Report for the
VEHICLE FOR SPACE TRANSFER AND RECOVERY (VSTAR)
VOLUME I**

A design project by students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of NASA/USRA Advanced Design Program.

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SUMMARY

The Vehicle Space Transfer and Recovery (VSTAR) system is designed as a manned orbital transfer vehicle (MOTV) with the primary mission of Satellite Launch and Repair (SLR). VSTAR will provide for economic use of high altitude spaceflight for both the public and private sector.

VSTAR components will be built and tested using earth based facilities. These components will then be launched using the space shuttle, into low earth orbit (LEO) where it will be constructed on a U. S. built space station. Once in LEO the vehicle components will be assembled in modules which can then be arranged in various configurations to perform the required missions.

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List of Symbols

A	-	area	ft ²
A _{tank}	-	surface area of fuel tank	ft ²
a	-	acceleration	ft/sec ²
a	-	semimajor axis	ft
a _{max}	-	maximum acceleration	ft/sec ²
a _{min}	-	minimum acceleration	ft/sec ²
F	-	force	lbf
g	-	value of acceleration due to gravity at earth's surface	ft/sec ²
g _c	-	gravitational constant	lbfm/lbf ft/sec ²
h	-	height	ft
h	-	altitude from surface of the earth	ft
h _{vapor}	-	heat of vaporization of fuel	BTU/lbm
ISP	-	ISP of engine	sec
K _{insul}	-	conductivity of insulation	BTU in./ft ² rankine
l	-	length	ft
m	-	mass	lbm
m _{air}	-	mass of air in the cabin	lbm
m _{brnout}	-	mass of vehicle at burnout	lbm
m _{cabin}	-	mass of cabin	lbm
m _{cargo module}	-	mass of cargo module	lbm
m _{crew}	-	mass of crew members	lbm
m _{dot fuel}	-	mass flow rate of propellents	lbm/sec
m _{evap}	-	mass of fuel that evaporates	lbm
m _{fuel}	-	mass of fuel required for given ΔV	lbm
m _{insul}	-	mass of insulation	lbm

$m_{initial}$	- mass of vehicle at beginning of burn	lbm
$m_{life\ support}$	- mass of life support systems	lbm
m_{pay}	- mass of payload	lbm
$m_{shielding}$	- mass of meteorite shielding	lbm
$m_{structure}$	- mass of structure of the cabin	lbm
$m_{support\ structure}$	- mass of the support structure for the fuel tanks	lbm
m_{tank}	- mass of fuel tank	lbm
m_{total}	- total mass of vehicle including fuel	lbm
n	- number of crew members	-
σ	- stress	psi
σ_{max}	- maximum allowable stress in a material	psi
p_{bottom}	- pressure at bottom of tank	psi
p_{fuel}	- vapor pressure of fuel	psi
p_{vapor}	- vapor pressure of fuel	psi
\dot{Q}_{dot}	- rate of heat leak into fuel tank	BTU/hour
Q_{total}	- total heat leak into fuel tank	BTU
R	- mass ratio	-
r	- altitude from earth's center	ft
ρ	- density	lbm/ft ³
ρ_{fuel}	- density of fuel	lbm/ft ³
ρ_{insul}	- density of insulation	lbm/ft ³
ρ_{metal}	- density of metal in structure	lbm/ft ³
$\rho_{metal\ in\ tank}$	- density of metal in tank	lbm/ft ³
r_{tank}	- radius of fuel tank	ft
t	- time	hr, min, sec
t	- time of burn	seconds
t_{insul}	- thickness of insulation	inches

T_{end}	-	endurance of mission	hours
Thrust	-	engine thrust	lbf
T_{in}	-	inside temperature of fuel tank	rankine
T_{out}	-	outside temperature of fuel tank	rankine
V	-	velocity	ft/sec
$V_{metal\ in\ tank}$	-	volume of metal in fuel tank	ft ³
V_{tank}	-	volume of fuel tank	ft ³
$V_{exhaust}$	-	effective exhaust velocity	ft/sec
ΔV	-	change in velocity	ft/sec
μ	-	gravitational mass parameter for earth	ft ³ /sec ²

DESIGN JUSTIFICATION

The industrial and commercial use of space has risen exponentially over the past two decades. Present indications are that this trend will continue long into the future. The industrialization and commercialization of space includes such activities as communications, earth resource studies, weather observation, materials processing, and energy production, just to name a few. These demands will result in an ever increasing need for low cost space transportation to geosynchronous and other high orbits. The increasing demand for these types of missions cannot be satisfied by the space shuttle due to its low orbit service ceiling. Thus the full potential of space commercialization can only be realized through the use of reusable space based OTVs.

VSTAR is a manned orbital transportation system designed to perform multiple missions while carrying out numerous tasks during each mission. The development and deployment of VSTAR will result in extensive technical and economic benefits to business, industry, and the military alike. At the present time geosynchronous satellites are deployed from low earth orbit using solid rocket boosters launched from the space shuttle orbiter. These systems are at present only partially satisfying the GEO system market. More efficient and powerful Inertial Upper Stage (IUS) systems are being developed and used; but again, these systems are ineffective in satisfying the commercial demand. Current systems have another disadvantage in that they are not reusable, nor can they repair, service, or in any way recover and bring back a previously deployed satellite. The use of these transportation methods results in large equipment and resource waste.

The development and use of a space based manned orbital transfer vehicle will eliminate many of the problems stated above. More efficient and economic geosynchronous operations can be achieved since the system will be completely reusable and will be capable of delivery, repair, refueling, and retrieving of geosynchronous and other high altitude systems.

I. INTRODUCTION TO YSTAR

The Vehicle for Space Transfer and Recovery will be designed as a modular Manned Orbital Transfer Vehicle (MOTV) (Figs. 1, 2). The vehicle will be capable of providing efficient and economic transportation to and from geosynchronous earth orbit (GEO) as well as assisting in other needed space operations. YSTAR will provide the capability to carry and/or retrieve two satellites to and from GEO. Other optional GEO operations include refueling, repair, and repositioning of satellites. A typical GEO mission for YSTAR will require a 2 to 3 day time period during which two new satellites are deployed, and two older satellites are recovered or repaired. YSTAR will utilize the best characteristics of state of the art space technology and incorporate advanced derivatives of past and present space vehicles. The fully reusable MOTV will be capable of carrying out varying missions and performing numerous tasks during each mission.

The crew for YSTAR will consist of three personnel including a command pilot, flight engineer, and payload specialist. The choice of a three person crew provides an optimum balance between mission capability, environmental control, and consumables. A three person crew also provides a safety factor for extra-vehicular activity (EVA). During EVA two astronauts will be needed outside the vehicle for retrieval and repair of satellites, while a third astronaut will remain inside to monitor the external activity as well as the onboard systems.

YSTAR's modular design will allow considerable flexibility in mission configurations thus adding to the economics of the system. Future versions of YSTAR will utilize advanced modules for more elaborate missions that can be added as demand and technology permits. The system consists of four primary modules which can be interchanged for performing different missions and tasks. The basic mission configuration for the SLR mission consists of command/control center, cargo bay, fuel storage, and propulsion modules.

The command/control module will provide vehicle control, monitoring functions and life support facilities. An equipment bay located behind the command module allows storage

of equipment such as manned maneuvering and robotic units along with other equipment needed for specific missions. The cargo bay module will be constructed using an open architecture truss assembly. This module will serve to carry satellites or other payloads as required by a given mission. The propulsion system chosen for VSTAR will consist of advanced chemical rocket engines with throttle, vectored thrust, and multiple restart capability. The propulsion module incorporates a central core engine assembly.

The highest contribution to operational costs for VSTAR will be the expense required to deliver fuel from the Earth's surface into LEO. To reduce these costs an efficient fuel/engine combination must be developed. Present research indicates that an advanced OTV propulsion system will provide the best performance. Although this type of propulsion system will require further research and development, the extra time and expense will be recovered during the mission life of VSTAR.

The development and deployment of VSTAR will add considerable impetus to space commercialization and industrialization. Numerous technical and economic benefits will result from such a system. This report provides a detailed look at manned orbital transportation systems and suggests a viable MOTV for use near the end of the century.

II. MISSION PLANS

It is essential that YSTAR be applicable to several tasks for it to be an economical design choice for an MOTV. However, the primary design considerations must be centered around the main mission goals, those perceived to be of greatest importance in the near future. The SLR mission will therefore determine the design criteria for the MOTV. The mission planning required for the successful achievement of this desired mission will be the deciding factor in such design aspects as payload capabilities and special servicing equipment as well as fuel, power, and life support systems.

The SLR mission is a complex task involving a combination of the manned maneuvering units and telerobotics needed to capture and load the satellites into the cargo bay of the MOTV. When immediate repair is not possible, YSTAR will be capable of retrieving two small satellites (approx. 2500 lbs each) or one large satellite (5000 lbs) and returning to the space station for more extensive repairs. In the same manner, YSTAR will be able to transport two satellites from the low-earth orbit of the space station (approx. 200 nm1.) to geosynchronous orbit.

A Hohmann transfer, involving an in-course plane change maneuver will be used for the trajectory from LEO to GEO. The Hohmann transfer (Fig. 3), consists of a two-impulse maneuver in which the main propulsion system is fired at the perigee of the initial orbit and then refired at the apogee of the target orbit to recircularize the orbit. After extensive study in the area of orbital mechanics, it has been determined that the Hohmann elliptical transfer will provide the best possible trade-off between low energy requirements and time constraints. The simplicity of the transfer method will also limit the maneuvering required for trajectory corrections.

Since the YSTAR will initiate the Hohmann transfer from the 28.5° inclination of the U.S. Space Station, a plane change will be required to reposition the MOTV in an equatorial

orbit (0° inclination). Though there are several possible means for executing this plane change, only the two methods which seemed most logical and economically feasible were investigated for VSTAR'S purposes. One method involves first transferring from LEO to a high-earth orbit (HEO) followed by a one-impulse plane change into GEO. The alternative method shown in Figure 4 involves a two-impulse maneuver in which the first ΔV is achieved at the ascending node of the LEO for a plane change of about 5.5°. The remaining 23.0° of plane change needed for GEO is then achieved through the second ΔV applied at the descending node of the transfer orbit.

Comparison of the two methods was based on the analytical calculation of the total ΔV required in each case. The results revealed that the LEO-GEO transfer involving a two-impulse plane change required about 20% less ΔV than the one-impulse plane change. The two-impulse method, therefore which requires a total ΔV of 14,271 ft/sec is more economical, requiring less fuel than the one-impulse plane change, requiring a total ΔV of 18,040 ft/sec.

The return trip to the Space Station will be executed by the same two-impulse plane change method, except the total ΔV will be achieved using reverse thrust of the same magnitude. To achieve the negative ΔV 's, the MOTV must be turned 180° about its yaw axis in order to position the propulsion module forward. This is needed since VSTAR contains only one main propulsion unit capable of producing the enormous pounds of thrust required for the LEO-GEO transfer. VSTAR's reaction control thrusters will be utilized to maneuver the MOTV into the reverse thrust position. As VSTAR approaches the node of orbit intersection, the primary reaction control thrusters will be used to generate angular momentum about the yaw axis. Both the primary and vernier thrusters of the reaction control modules will then provide the fine attitude adjustments to realign the MOTV on course.

Though the design specifications of VSTAR are based on the SLR mission requirements, the MOTV will be capable of performing various other tasks. As a space-station based vehicle, the MOTV will also be available for emergency situations involving spacecraft

with system failures or mission complications. One much needed operation in which VSTAR is suited is the collection of "space garbage". As more and more communication satellites are developed each year, the need for removal of space debris increases. With the satellite retrieval systems on board, VSTAR will be capable of transporting space debris to higher earth orbits or back to the space station.

One closely related application for the MOTV is the Interplanetary Vehicle Assist (IVA) mission which was considered during the preliminary design configuration of VSTAR. The IVA mission is a new concept that was perceived to be an excellent alternative in making the plane change necessary for repositioning an interplanetary spacecraft (ISC) to an ecliptic orbit. To execute the mission VSTAR would be connected to the ISC using a rigid linkage system consisting of hydraulic braces which permit spin, pitch, and yaw control through thrust vectoring. It would serve as the main propulsion system by providing the ΔV for the plane change as well as working in conjunction with the ISC's attitude control thrusters (ACTs) to provide trajectory corrections. Since the ISC of the near future will most likely be a space-based vehicle assembled on a U.S. space station in a 28.5° inclination low-earth orbit, the plane change to place the vehicle in the ecliptic plane will require large amounts of fuel for producing an estimated total impulse of approximately 22,504 ft/sec. Further investigation into the IVA mission has however produced some discouraging results in the practicality of the mission. Based on an assumed ISC weight of 800,000 lbs, the fuel requirement in executing the plane change maneuver for the combined weight of the ISC and VSTAR is estimated to be 13 to 14 times greater than the designed fuel capacity of VSTAR. This means that several sets of strap-on tanks would have to be added to provide the increased volume of fuel. Though the mission would significantly decrease the total weight of the ISC, this would only create more fuel expense and possible structural failures for VSTAR.

Other complications including engine performance and the Vehicle Interface Assembly (VIA) used for linking the two spacecraft, are major factors considered in the analysis

of the IVA mission. The plane change maneuver for the combined system would require more than double the normal thrust of the MOTV's propulsion system. Maintaining rigidity in the linkage system limits the acceleration that the system can withstand and therefore higher burn times are required of the engines. Since engine life expectancy is directly proportional to the burn time of the engines, using VSTAR's engines for the IVA mission would therefore lower the operating life of the vehicle. Storage of the VIA would also introduce complications in the mission plans. The preceeding reasons come from careful analysis of the mission elements and provide supporting evidence for the decision that the IVA mission will not be feasible under near-future technology.

The propulsion, power and life support systems are selected through careful consideration of the amount of fuel and time required for each mission. The total mission time anticipated for the SLR mission is two to three days depending on the particular repair operations required. This includes the estimated 10 to 11 hours needed for the round-trip trajectory. Based on this consideration, the propulsion system used in VSTAR must have as high a specific impulse (Isp) as possible. Furthermore, the fuel capacity needed for a three-man crew and the life support and power capabilities of the MOTV must surpass the required level of the SLR mission in order to comply with safety standards.

Mission planning for the SLR mission must also incorporate the travel constraints introduced by the space environment. Solar flares and Van Allen radiation belts release large amounts of radiation which can damaged the MOTV systems and cause fatal injury to the crew if time of exposure or radiation dosage is too high. Since VSTAR will be primarily designed for short term missions, extensive pre-flight planning will prevent the need for additional radiation shielding on the external structure. Including the use of Van Allen radiation belt charts and solar flare predictions, in the planning of the VSTAR mission, will provide accurate determination of a safe mission schedule.

III. GUIDANCE AND CONTROL

In order to accomplish the SLR and other MOTV missions, the guidance, navigation and control (GN&C) subsystem must satisfy a host of mission and performance requirements. The system must provide the capability to rendezvous and dock with other spacecraft and the space station. It needs also to provide the capability to de-orbit payloads, to deliver satellites to precise orbital conditions, and to support extended in-orbit operations such as satellite repairs.

To provide the attitude control for VSTAR, the best available sensing and data processing hardware will be incorporated into the GN&C subsystem. Since guidance and navigation is based on determination of the position and acceleration of the spacecraft, on-board sensors will be used to gather data for the GN&C computer in order to determine the craft's orientation and velocity. An inertial reference unit will be required to provide continuous attitude knowledge information to the GN&C function. Two Inertial Measurement Units (IMU's) located forward of the flight deck will measure the accelerations of the MOTV and convert them into position vectors. The IMU's will be re-aligned using startrackers and crew optical alignment sight (COAS) via GN&C processor commands.

Startrackers are optical devices which sight several bright navigational stars and transform their location into a vector. Using the GN&C processor, this vector is then compared to the star's known vector to determine the position of the craft. There will be two startrackers aboard VSTAR, located in the front lower section of the flight deck. One will be aligned in the direction normal to the side of the MOTV and the other in the direction normal to the top of the craft. There are two types of self-contained startrackers, Boresighted and Gimbaled, that were considered as candidates for the MOTV GN&C subsystem. Studies of both sensors resulted in the decision to equip VSTAR with the Gimbaled Startrackers. This choice is based on the perception that the simplicity and minimum size and weight of the boresighted tracker won't provide advantages comparable to the flexibility offered by the gimbaled tracker. This

flexibility comes from the fact that its field of view is not limited as is the case with the boresighted tracker (Ref. 26).

COAS is simply a manual version of a startracker which will be used to sight stars through a cross-haired scope in the back window of the MOTV flight deck. This system of navigation works by recording the time of crossing of a particular star and creating a position vector. The data then can be compared to the IMU obtained position in order to more accurately align the IMU's with the startrackers. Rate gyros and accelerometer assemblies will also be implemented into the operational design of YSTAR as a secondary guidance system. Location of these components will be based on that which optimizes sensing capabilities. The rate gyros will assist the IMU's in measuring the angular and translational velocities of the spacecraft. A radar tracking sensor will also be included in the GN&C subsystem to support spacecraft rendezvous and docking operations into the visual piloted range. In order to safely perform operations in the dark, lights will be utilized to illuminate the target for visual piloted control operations (Ref. 26)

The space station will provide the navigational reference needed to initialize the position of the craft. Once this data is known, the MOTV's sensors will determine the craft's location and motions. The attitude control operation process will begin once the orientation and velocities are determined by sensors (Fig. 5). The signal will then be fed into the GN&C Data Processing Computer, which transfers the signal to the CRT displays in the crew command module. A pre-programmed or a piloted maneuver signal is then transferred back through the computer to the reaction control thrusters. The reaction control system will then use the optimal engine firing pattern to execute the maneuver.

IV. PROPULSION

System Function

The need for reusable systems for transporting space hardware between low and high energy orbits will steadily increase over the next few decades. The extension of manned operations from LEO to GEO is naturally the next step in man's conquest of space. These types of missions will require a MOTV capable of fulfilling a multitude of roles and tasks. The propulsion for such a vehicle is crucial to both its technical as well as economic success.

The propulsion subsystem must be able to accomplish the mission within a specified time frame while maintaining a high amount of efficiency. The complete system must be capable of providing both primary and auxilliary propulsion. The primary propulsion system provides the main propulsive thrust for mission accomplishment while the auxilliary propulsion system provides attitude control and low thrust maneuvering capability. The primary propulsion system is responsible for performing the following operations.

- Insertion of VSTAR into transfer trajectory between LEO and GEO.

- Circularization and injection of vehicle and cargo into GEO.

- Orbital maneuvering within GEO to deploy and pickup cargo or other mission assignments.

- Injection of VSTAR into transfer trajectory between GEO and LEO.

- Circularization and injection of VSTAR and returning cargo into LEO.

This section provides a description of the propulsion considerations and final selections made for VSTAR along with the justification for each final choice. Topics include the general propulsion design requirements, the types of propulsion systems considered, and a detailed description of the system selected.

Design Requirements and Guidelines

The primary factors in the propulsion system design include thrust, mission time, propellant type, reliability, and the availability of technology by the proposed operational date. The requirements used to define the operating constraints of the primary propulsion system are listed below along with their rationale.

High Specific Impulse (Isp) - The general requirements and guidelines for the MOTV were based on the need to minimize the amount of propellant required while maintaining optimum performance and crew safety. The need for this low propellant requirement is justified in light of the fact that the highest cost of the VSTAR project is the delivery of propellant from earth to LEO. Even the smallest savings in fuel will result in large savings over the ten year life of the vehicle. Careful analysis of several engine/fuel combinations has shown Isp to be the greatest contributor to the reduction in propellant. For this reason the engines used on VSTAR must possess as high an Isp as possible.

Reliability - High system reliability is required due to the fact that this is a manned vehicle.

Service Life - Each engine must have a long service life with minimum maintenance in order to reduce overall operating costs.

Control - Each engine must be capable of multiple restarts over numerous missions prior to the need for engine overhaul. The capability for thrust vectoring and throttling of each engine is also required to allow flexibility in mission profiles and cargo while at the same time allowing redundancy so that the system can maintain operation in the event of an engine failure.

Propulsion System Candidates

Numerous types of propulsion systems suitable for a manned system have been carefully

examined to determine their feasibility to provide primary propulsion for VSTAR. These types include electric, nuclear, laser, and chemical systems. Each of these basic types is briefly described below.

Electric propulsion - There are three basic types of electric propulsion currently being developed. These types include electrostatic (ion) propulsion, electrothermal (arcjet, resistojet, and microwave) propulsion, and electromagnetic propulsion. Electric propulsion systems provide very high specific impulse typically in the range of several thousand seconds. This high I_{sp} makes them a likely candidate from an economic point of view. However, these systems require extensive power systems and are usually limited to low thrust levels. Typical acceleration rates for electric propulsion systems are on the order of 10^{-4} g's or less (Ref. 21). Such low accelerations would require a spiral trajectory for a mission from low Earth orbit to geosynchronous orbit resulting in a transition period of several months. The primary mission of the MOTV will necessitate traveling through the Van Allen radiation belts. The time period for the transition through this region must be kept as short as possible to limit crew and system radiation contamination. A long transfer period is impractical for a MOTV from a life support and safety point of view. For these reasons electric propulsion was ruled out as a candidate for VSTAR propulsion.

Nuclear Propulsion - Nuclear propulsion systems use low density fuel such as hydrogen without the need for an oxidizer. The fuel is superheated by a nuclear reactor and expanded through a nozzle to produce thrust. Nuclear propulsion systems have very high I_{sp} ratings on the order of 700 - 2800 seconds, making these engines very economic while providing a large thrust (Ref. 21). However, there are disadvantages to these systems as well. Nuclear reactors are as yet unsafe and require thermal and radiation shielding which adds to the mass of the

vehicle. The operating engines also cause trapped radiation contamination in and around inhabited areas like the space station. Nuclear propulsion may prove to be a viable candidate for future MOTV systems; however, these systems still require considerable development and are unsuitable for YSTAR at this time due to the disadvantages mentioned.

Laser Propulsion - Laser propulsion systems utilize permanently based high power lasers to superheat the fuel carried in a target vehicle. The superheated fuel is then expanded through a nozzle to produce very high thrust. The Isp ratings of these systems are expected to be comparable to those of nuclear propulsion systems (Ref. 21). The main advantage of laser propulsion is the savings in weight due to the fact that the primary combustion producing source (the laser) is located away from, and is not actually part of, the vehicle itself. Although great strides have been made in laser technology, there are currently no high power lasers nor associated power sources capable of accomplishing this task. The avionics for target tracking and guidance also prove to be a major problem. As can be seen the technology for laser systems still requires much development and will not be available for some time. The use of laser propulsion for YSTAR is therefore not a viable option.

Chemical Propulsion - Chemical propulsion systems fall into two major categories, solid and liquid, each of which produce thrust by combusting a fuel and oxidizer mixture and expanding the hot exhaust gases through a nozzle. Both types are currently being used to provide spacecraft propulsive power, and have an enviable performance record. Solid chemical propulsion systems use a solid propellant (fuel and oxidizer mixture). These engines have very high thrust levels due to the high propellant density. However they are very hard to control and have limited restart capability. Solid engines also have low specific impulse on the order of 200-300 seconds (Ref. 18). For these reasons solid propellant systems are not suitable for YSTAR.

Liquid chemical rocket engines utilize separately stored liquid fuel and oxidizer and come in a variety of configurations and operating modes. They may be used for either low or high thrust operations and have a reliable thrust control capability. Other advantages of liquid propulsion systems include built in open loop cooling systems, and most importantly availability of technology (Ref. 18). The greatest disadvantage of liquid systems is the large propellant requirement. However, recent advances in liquid chemical propulsion technology will allow a wide range of variable thrust and higher Isp ratings (400-500 sec) which will help reduce the amount of propellant needed.

Advanced Technology Considerations

One method of reducing the propellant requirement while maintaining payload capability and high engine performance is to utilize new concepts which are currently being studied for future systems. Several different advanced concepts were studied in hopes of optimizing VSTAR propellant use as much as possible. Concepts studied include the use of both the aerobrake and the dual expander and dual-fuel/mixed mode engines.

Aerobrake - Aerobraking uses the aerodynamic forces of the earth's atmosphere to slow down a returning spacecraft so that it can enter a low earth orbit. This technique effectively eliminates the need for a propulsive return ΔV maneuver required to circularize the orbit around the earth. Aerobraking has the advantage that it saves large amounts of propellant and mass. However, aerobraking also has its disadvantages. The equipment needed for aerobraking consists of a large ballute or umbrella type shield that deflects heat away from the vehicle during reentry. The system also requires a thermal covering to protect the vehicle from residual heat. Even if made from very thin material these shields and covers are extremely heavy. Another major disadvantage of aerobraking is the fact that the technique has never been tested and is based solely upon theory at the present time.

After examining the aerobrake concept it was decided not to use an aerobrake for VSTAR due to the disadvantages mentioned as well as the structural design and mission of the vehicle. The chosen structure of VSTAR limits its ability to support the high aerodynamic loads which surely accompany aerobraking. The open architecture design also provides no thermal covering to protect a returning payload during the aerobrake maneuver. To utilize aerobraking for VSTAR a reentry configuration must be developed consisting of a stronger more massive structure protected by the main shield as well as by thermal insulation coverings. Any arrangement of shield and structure will result in excessive weight and structural problems. If placed at one end of the vehicle, less surface area of the vehicle itself must interact with the aerodynamic forces occurring during reentry but the required size of the shield grows exponentially with the required rearward distance that the shield must envelope. If the shield is placed along the length of the vehicle the size is reduced only slightly and even more vehicle surface area is now interacting with the aerodynamic forces, so the structure must be even stronger. The added weight, expense, and maintenance of the shield and its support, as well as the added weight and cost needed to increase the strength of the vehicle structure makes aerobraking undesirable at this time.

Dual Expander and Dual-fuel/Mixed-Mode Engines - An engine that uses the dual-fuel / mixed-mode concept burns a tripropellant combination of two fuels and one oxidizer. This concept usually consists of two fuel and oxidizer combinations. One combination (mode 1) consists of both a high density, low Isp hydrocarbon fuel and oxidizer (LOX/RP-1). The second combination consists of a low density higher Isp fuel and oxidizer (LOX/LH₂). Both combinations are burned in the same stage. The combustion of these two propellant combinations (modes) can be done in sequence or in parallel thus allowing fuel usage to be tailored to a particular mission. This mixed mode principle benefits some vehicles by decreasing the

propellant mass and volume as well as the overall propulsion system structure. The mixed-mode system can be incorporated by using entirely separate rocket engines for each mode or by using a dual expander engine.

The dual expander - variable throat engine concept (Fig. 6) allows either a bipropellant or dual-fuel/mixed-mode (tripropellant) system to burn the propellant in a double combustion chamber arrangement (Ref. 5). The system consists of both an inner primary combustion chamber surrounded by a secondary chamber. The dual chamber system operates at higher chamber pressure values than that obtained in single chamber systems, thus improving the Isp for a given propellant combination. The dual chamber arrangement also has a higher expansion ratio which typically reduces the nozzle expansion bell housing by nearly a half of that found for single systems (Ref. 5). The design also incorporates a variable throat. The variable throat allows adjustment of the nozzle area ratio thereby providing near optimum performance at all thrust levels while maintaining the same nozzle exit area (Ref. 5).

Several trade study analyses were conducted to compare conventional bipropellant (LOX/LH₂) systems with dual-fuel/mixed-mode systems for given mission and vehicle parameters. Figure 36 shows the propellant required for a given Isp rated engine incorporating either a bipropellant or a dual-fuel/mixed-mode system. The results indicate that better propellant efficiency can be achieved using one of the advanced orbital transfer vehicle engine designs modified to include dual expander - variable throat technology. No appreciable savings were found for VSTAR using the dual-fuel/ mixed-mode concept. This result was primarily due to the lower Isp rating that is characteristic of most hydrocarbon fuels such as kerosene or RP-1.

It should be pointed out that these studies are in no way exhaustive but rather represent the best choice from currently available technology. It is also important to recognize that improvements in both the design as well as the propellants of dual-fuel/mixed-mode systems

may increase the I_{sp} ratings of these engines within the next 10 - 15 years. Figure 3@ shows a theoretical case for a dual-fuel/ mixed-mode engine with a high I_{sp} (492 secs.) which indicates that if such an engine could be developed it would require slightly less propellant and therefore should be used for VSTAR.

VSTAR Primary Engine Selection

Several existing and developing engines were considered for the primary propulsion system. Most of these systems were ruled out due to low specific impulse, no restart, or no throttleable thrust capability. A few engines do, however, fit the specified propulsion guidelines. For the reasons previously discussed an advanced OTV chemical propulsion system has been chosen as the primary propulsion for VSTAR.

The main engine assembly selected for VSTAR will consist of three Rocketdyne Advanced Orbital Transfer Vehicle engines (Fig. 7). The engines will be modified using the dual expander - variable throat design, thus improving the I_{sp} while reducing size and weight. The chosen engine does not yet exist as an off the shelf item but is currently under development and should be available within the time schedule proposed for VSTAR operation. Each engine will be fully throttleable and gimballed to provide vectored thrust capability. A three engine cluster has been selected to provide a balance of mass, cost, performance, engine life, and redundancy. Together these engines will be capable of providing a velocity change maneuver on the order of at least 28000 ft/sec in order to accomplish the geosynchronous mission.

VSTAR Propulsion System Design

Three primary MOTV propulsion system configurations have also been considered to determine the most cost effective design for VSTAR. These include the single stage, 1 1/2 stage, and common stage (Fig. 8).

Single Stage - The single stage system is the simplest but results in excess burn-out mass as the propellant is consumed. This design does however offer the added advantages of simplicity and reusability.

One and One-Half Stage - The one and one-half stage system uses a single core engine system with externally mounted drop tanks which are released as the fuel is used up, much like the space shuttle external tank. These drop tanks effectively increase the performance of the system, but are inefficient since the used tanks are not recovered and therefore continually add a nonrecoverable cost. Another configuration using reusable drop tanks at first appeared to be a viable solution to this problem. However these tanks require their own propulsion systems and avionics to return them to a parking orbit in LEO, thus negating the advantage of their use.

Common Stage - Common stage systems provide a compromise between mass economy and reusability. These systems utilize a staging process whereby each stage is dropped as its fuel is consumed. The empty stages are then returned to LEO where they are refueled for future missions. Although they appear to be efficient, common stages have a drawback in that each stage requires a separate engine, as well as avionics equipment for the return trip to LEO, thus negating the original fuel and weight savings.

Of the three systems evaluated a single stage configuration was finally chosen as the best design for YSTAR since its simple design ultimately provides the best cost of the three systems studied and also limits the overall problems encountered (Fig. 9).

Auxilliary Propulsion System

The reaction control system used on YSTAR will allow attitude control (pitch, roll, or

yaw) and positioning of VSTAR near target spacecraft by performing translational and angular speed changes. There are two primary means of providing reaction control - angular momentum devices and thrusters. Angular momentum devices include reaction wheels, momentum wheels, and control moment gyros. In order to provide the control required for a vehicle the size of VSTAR these momentum devices would have to be massive. Also, angular momentum devices provide only rotational control so that thrusters are still required to provide for translational maneuvers. Thus the best method of providing reaction control and vehicle maneuvering for VSTAR is in the form of an auxiliary propulsion system utilizing primary and vernier thrusters.

The choice of the auxiliary propulsion system used on VSTAR is based on current state of the art technology. A system similar to that used by the space shuttle has been selected due to its availability and proven reliability (Ref. 25). The auxiliary propulsion system will consist of one forward module and two aft modules, each with its own monomethyl hydrazine fuel and nitrogen tetroxide oxidizer storage system (Fig. 10). The primary and vernier engines selected for VSTAR are the Marquardt R-40A and R-1E respectively (Fig. 11). Twelve primary thrusters are used for each module. The primary thrusters are each capable of producing about 870 pounds of thrust and will be used to provide normal translational and rotational control. Four vernier thrusters are also provided on each module, each capable of 25 pounds of thrust. These vernier thrusters will be used to provide fine adjustments in vehicle attitude and position.

V. STRUCTURE

Configuration

The configuration chosen for VSTAR will enhance mission flexibility and lends itself well to future improvements in spacecraft technology. The structural components of the MOTV will be four separate modules linked in the following order: command/control cabin, cargo bay, fuel storage module, and the propulsion module. The modular design will allow for simple repair and maintenance in space and will permit vehicle adaptation for various mission requirements. When the engines require overhaul, they can easily be disconnected and temporarily replaced while extensive maintenance is performed at the space station or on Earth.

The command/control cabin will be connected at the front of the vehicle and will contain the vehicle control center and the living/mission section which enables the crew to have a comfortable work area during their mission. This cabin will contain an airlock for EVA and cabin access. Power generation systems, communication/data link systems, computers, flight control systems, life support systems and mission systems will be housed in this module. The command/control cabin will have radiation and thermal control to allow human and electronic habitation with sufficient protection from debris penetration. The size of the module is designed to have cylindrical shape with a diameter of 13' and a length of 23'.

For the MOTV to accomplish its primary mission, it must have some means of transporting cargo. The base design cargo module is a light truss structure that can carry 9000 cubic feet of cargo. The diagonal supports on the top side of the bay will be retracted for satellite docking in and out of the bay. The maximum mass of the cargo depends on the particular restraints and vehicle accelerations, but for the primary mission of transporting satellites from LEO to GEO, the payload limit is 5000 pounds. The SLR mission also demands that the payload area contain a telerobotic arm for retrieval and deployment of satellites. A telerobotic work station (TWS) will also be located in the cargo bay for performing various repairing and refueling operations and

minimize the EVA of the crew. The size of the payload area is therefore determined by the area needed for servicing. The planned dimensions of the bay are 45' X 16' X 16'.

The only purpose of the fuel storage module is to store the fuel of the vehicle. This section being modular will allow mission planners more flexibility for future missions as new developments in fuels and fuel storage are discovered. The currently proposed design is optimized for the primary mission and any deviation from this will not minimize the fuel requirements. The fuel section permits the stable storage of cryogenic and multi-fuel systems. One radiator/heat pipe system will be allotted for the combined fuel and propulsion section to minimize boil-off and thermal conduction from the engines. The funnel shape of the proposed fuel section will have a maximum diameter of 15', a minimum diameter of 7' and be 53' in length.

The propulsion module contains engines and the thermal control unit previously mentioned. The modular design allows for maintenance simplification, as well as providing designers with a simple means of improving MOTV performance as new engines are developed. The estimated dimensions of the propulsion section will be 7' in diameter by 6' in length.

Thermal Protection

Thermal protection for the crew and the structure is of prime importance in the success of the mission. Heat is generated within the vehicle from several sources, including the engines, electrical systems and crew. There is also a significant flux of heat from the Sun and Earth. Insulation, heat pumps, radiators and heaters are the devices which will control the heat flow through the MOTV.

Multilayered insulation (MLI) is a standard type of insulation popular in today's spacecraft. MLI reduces the flux through the interior of the spacecraft by redirecting the heat flow around the vehicle rather than through it. This insulation has proven its effectiveness in many satellites and should do the same for YSTAR. Another insulating measure will be the paint of the surface exterior of the vehicle which will reflect much of the solar radiation. Heat pumps, radiators and heaters will be connected together in a control system to allow a stable environment

for the interior of the MOTV. This system will also be a part of the environmental control system needed for the human occupants. It is designed to handle a maximum heat flow into or out of the interior of the vehicle from the external and internal environments.

The thermal control system components of this vehicle will be similar to that of the STS. They will however be scaled down due to the smaller crew size and the absence of aerodynamic heating from atmospheric re-entry. Since VSTAR never enters the atmosphere, the only heat sources are the internal heat of the vehicle, the heat provided by the Sun, and the albedo of the Earth. The solar output is 408 to 451 Btu/ft²-h the earth's radiation is 72.9 to 77.4 Btu/ft²-h and the space sink temperature is 0 degrees R. (Ref. 7). The hotter case will occur when VSTAR is between the earth and the sun. At that time the area facing the sun and the earth are equal at 420 ft² and the external heat input is 221,992 Btu/h. Internally, the electronics and power generation will input a maximum of 10,000 Btu/h, the crew input around 3,000 Btu/h. An isolated thermal control system for the engines and fuel tanks will be developed to handle the total heat transfer. The cargo bay will be thermally connected to both the cabin and the fuel section but the heat input from this section is minimal and would be limited by the structural material. The cabin is estimated to have an external heat flux maximum of 153,940 Btu/h and an internal input of 13,000 Btu/h. The MLI allows a net inflow of 1,522 Btu/h added to the 13,000 Btu/h that remains inside the vehicle for a total of 13500 Btu/h that must be removed by the radiator. The maximum heat input to the fuel and engine modules occurs at the same position in space with the engines operating at maximum thrust.

The choice of the type of heat exchanger to space is the key to limiting the weight of the thermal control unit. The radiator used will be a rotating bubble membrane type that cools the fluid by spraying it outward into an enclosed rotating sphere which then collects the fluid at the center line of the sphere for reuse. The surface area for the cabin radiator sphere must be 25.83 ft², giving it a radius of 1.66 ft and a total system mass of about 50 lbm. for the required output. (Ref. 31). An environmental heater will also be placed in the command/control cabin to avoid overcooling which may occur at any time other than at maximum design input heat flux. A more

accurate analysis can be obtained by the use of Finite Element Analysis for the desired minimum and maximum conditions.

Additional thermal and radiation control will be applied to the three optical view ports in the command/control cabin. Although the vehicle would be better at protecting the crew without the thermal leakage and radiological acceptance of windows, visual capability will be helpful in docking and in locating satellites or non-signal-transmitting objects. VSTAR will have two windows in the cockpit section and a single window facing the cargo bay to provide the mission specialist with a physical view of the bay and facilities. Each window must be constructed to minimize the undesirable effects. The methods for this have been refined over the years of space exploration. The process is relatively simple but the product expensive. A special glass is produced containing a small percentage of Iron Oxide, which limits the passage of harmful ultraviolet radiation. To prevent penetration of excessive thermal radiation, a thin layer of gold is used as the sandwiched layer between two plates of glass. A thin layer of aluminum oxide and magnesium fluoride are applied to the exterior surface of the window to limit X-ray penetration and unwanted surface reflection, respectively. The design of the windows was first used by earlier spacecraft including Apollo and will be utilized by VSTAR.

Materials Selection

Although the limiting factor in determining structural weight is the extensive debris and micrometeoroid protection, the loading of the structure due to acceleration is also analyzed to determine material requirements. To minimize the thickness and mass of the structural walls the material with the highest specific strength will be used. For composites, this specific strength is dependent on the direction of the applied force. Cost of the material production is another important factor in material selection. Cost per unit mass of a material multiplied by the mass required for that particular material to support the load determines the cost of the material. Aluminum and steel have very low costs while composites are relatively high priced. Compatibility with the environment and with other materials is also a factor in determining the

best materials for a function. Space is a harsh environment, extremes of heat and cold, the absence of any external pressures and destructive radiations. It quickly causes an exposed material to deteriorate. Corrosion due to internal fluids and contact with dissimilar materials also weaken and destroy materials. Aluminum deforms under the temperature variations and epoxys dissolve relatively quickly in the emptiness of space. Aluminum will not survive long when it contains corrosive fluids and steel in contact with aluminum will corrode.

After material analysis, aluminum is determined to be the best material for the debris shielding since it is inexpensive and under no structural loading. Material selection for the other parts was more complicated. For the command/control cabin, the predominant structural material will be a Boron Aluminum composite. Its strength and the fact that it is a metal composite give it better characteristics for debris protection. Steel has been chosen for the fuel tanks since the fuels have relatively high pressures and corrosive tendencies. Because of the size and required strength of the cargo section, a Graphite Epoxy composite with an aluminum casing has been chosen for this module.

Debris and Micrometeoroid Protection

The lifespan of the vehicle will determine the cost effectiveness of VSTAR. The longer the vehicle can perform its mission, the less the cost per mission. In space, there are a number of events that can limit the life span of a spacecraft, but debris and micrometeoroids can inflict the greatest structural damage. These small particles which are travelling in orbit with the vehicle can impact with it and causing a great deal of damage by penetrating pressure tanks or shattering support beams. In the low altitude orbits, such as VSTAR's parking orbit, the debris is relatively dense due to man's space exploration. For this reason VSTAR must be shielded.

To obtain a life span of ten years requires a large amount of shielding. VSTAR's design has opted for a 99.0% chance of not having debris or meteoroid penetration for ten years. As compared with NASA's requirement of 99.99% protection percentage over ten years for the planned space station, VSTAR's percentage seems a little low. However, unless there is a great

increase in the need for satellite repair and fueling, VSTAR will be spending most of it's life unmanned in a low-earth orbit. This results in a very low percentage chance that debris or micrometeoroids will penetrate the command/control cabin, fuel tanks or structure of the spacecraft during the more critical, manned periods of VSTAR's lifespan.

Using the NASA program, BUMPER, the minimal amount of shielding was determined to give the desired 99% protection for the ten year duration. This program calculates the survivability percentage based on duration, shielding and wall thicknesses. Optimizing for the minimal thicknesses results in the lowest amount of weight addition required for adequate protection, thus minimizing the additional fuel and cost required. BUMPER was used to determine the survivability of each module. The results can be seen in table 1, and a graphical representation shown in figures 12 & 13.

Table 1. Results of BUMPER Analysis on VSTAR

Module	shield wt. (lbm)	stand off dist (in)	shield thickness (in)	MLI
Cabin	286.05	16	0.0156	yes
Cargo	162.77	3.5	0.0260	no
Fuel Tanks	116.20	5*	0.0208	yes

* added to the average of 6" from tanks

The cargo bay is an unusual case in that if the vessel is penetrated there is no fluid loss. However, because an epoxy was chosen as the structural material a penetration could mean that the beam shatters losing all of it's strength. An arbitrary total radial limit of 3 inches total was allowed for the beams of the cargo bay to ease handling of the cargo, simplify module construction and minimize the surface area of the structure. To minimize the shielding weight for the tanks, a single shield can be placed around all the tanks. An assumption of BUMPER is that the debris

impacts normal to the surface. In the particular case of the fuel tanks, particles impacting normal to the surface is improbable, due to the geometry of the structure. An average distance from the shield surface to the tank wall is assumed to be the tank wall in the program to improve the accuracy of the output. Due to shuttle transfer requirements for the vehicle, an armor offset distance limit must be accounted for the command/control cabin shielding.

With the micrometeoroid shielding, the load bearing members are now protected from excessive wear and abrasion. However, communications equipment and heat pipes as well as all other exposed devices will slowly be worn away due to debris, micrometeoroids, and evaporation of the materials into space. These devices will have to be covered externally or stored within VSTAR, while not in use. Much of the communication equipment will be housed within the MOTV to avoid breakage during satellite loading, unloading, and servicing. The remaining devices and the entire external portion of the vehicle should be painted with thermally reflective paint and an additional thin film of polyurethane. This coating will have to be reapplied every few years to compensate for evaporation and sandblasting from debris. This should significantly extend the life of the external devices, the meteoroid shielding, as well as the entire vehicle.

The thermal control, materials selection, micrometeoroid and debris protection and required strength are all combined in construction. Cross-sectional views of VSTAR's structural elements can be seen in Figure 14. These methods of construction minimize weight and maximize performance of the thermal control system and shielding. Module joints and cargo bay connections are also designed for ease of operation and construction in space, while allowing for high stresses and loads. The joints function by interlocking the connectors and screwing the locking mechanism over the interconnection, (Fig 15). This method is currently being studied for space construction by several companies. These joints must also allow power and data communication lines to run to the propulsion section from the command/control cabin. This is accomplished by the use of standardized electrical connectors at the joints which allow the lines to run parallel to the structural members.

VI. LIFE SUPPORT

The Environmental Control Life Support System (ECLSS)

The ECLSS used on the VSTAR is the system that provides for the comfortability of the crewmembers. This system regulates the temperature, pressure and water supply, and provides the facilities for sleeping, storage of food, and waste collection. In other words, the ECLSS is the system that will keep life onboard VSTAR as comfortable as possible. The ECLSS will control cabin life for three crewmembers- pilot, flight engineer, and payload specialist.

Specifically, the ECLSS will take into consideration the following:

- a. Atmospheric revitalization system
 - Control of the temperature, cabin pressure, humidity
- b. Facilities (cabin design)
 - Vertical Sleeper /Waste Collector /Food Gallery
- c. Water and Food Supply
- d. EVA Support

Many of the systems employed by the MOTV are similar to the control systems used in the Space Shuttle, however the size is optimized for VSTAR.

Atmospheric Revitalization System (ARS) - To provide the proper atmospheric conditions, oxygen must be replenished and the harmful gases eliminated. The standard conditions that are needed for the ideal atmosphere in the cabin are: (Ref.29)

Air Temperature- 16 - 26 ° C

Atmospheric Pressure- 14.7 psia

Atmospheric Comp- 21% Oxygen, 79% Nitrogen

Humidity- 5 - 16 ° C (dew point)

One of the jobs of this revitalization is to remove traces of contaminants as well as CO₂ from the cabin air. To do this, filters will be used that contain Lithium Hydroxide (LiOH) and activated charcoal, which absorb the CO₂ and remove the contaminants. These filters are replaceable and can be easily maintained. The second job of this system is to maintain the standard conditions which were previously stated. This is done through the computer system that monitors and controls the temperature, pressure, humidity, and other conditions. The computer system is the main regulating device in YSTAR, but at anytime the program can be overridden for personal preference.

The Facilities of Comfort - The facilities available on YSTAR will be designed and installed with the comfort of the crew in mind. The main facilities of YSTAR are the vertical sleeper (VS), the urine/fecal collector, the food galley, and the personal hygiene center. There will be only two sleepers available in the cabin since one crewmember will be on duty at all times. The VS (Fig. 16) will have all the requirements needed for each crew member such as crew preference kits, trash containers and bags. Since there are only two sleepers for the three crewmembers, the personal storage compartments will be located elsewhere in the cabin. The sleepers will be retractable to increase the open area space when they are not in use. The sleepers are vertical to minimize the space needed for sleeping. In the absence of gravity, sleeping position will not matter. Straps are located in the sleeper to stabilize the crew member's body while asleep.

The second major facility is the waste collector which collects and disposes of both liquid and solid waste. This collector (Fig. 17) will be similar to the one used in the Skylab missions. The collector may be larger than the one used in the Orbiter; but with the cabin design, size is not a constraint. Also, since the Orbiter had problems with the reliability of its waste collector, it seems much more sensible to use the one in Skylab.

In the early flights of space travel, many different types of food containers have been

tested. They ranged from the metal, squeezable tubes and spoon-bowl packaging of the Apollo missions to the freeze-dried foods of Skylab. For VSTAR, a combination food gallery/hygiene center (Fig. 18) will be used. This food gallery is the third major facility in the cabin. The gallery will be located towards the aft end of the cabin. This gallery is a cabinet-style gallery that consists of serving trays, a pantry, hot/cold water dispensers, and a conventional oven for warming food. The gallery has two doors that open up to reveal a fold out table for food preparation. Many types of food will be stored in several different ways in the pantry including dehydrated and freeze-dried food. The beverages onboard will consist of instant mixes which are added to water. Examples of these beverages include tea, lemonade, Koolaid, and Tang.

Water and Food Supply - Since the VSTAR is a manned OTV, food and water supply needed for an average mission had to be taken into consideration. Water will be used for coolants, personal hygiene, and drinking so an average of 8 lbs of water should be needed per man per day; about 6.5 lbs is for drinking. The types of foods will range from mission to mission, but the amount that is needed will remain relatively the same. There are many items that are considered to be consumables (items that are non-recyclables) and these items will need to be replenished for each mission.

Interior Cabin Design - The interior design (Fig. 19) shows the location of all the facilities that will be used in the VSTAR from the sleeper to the control station in the rear of the cabin. To best show this, two views were drawn so that the left and right side of the command/control module can be seen. Other views of the cabin are an overhead view and side view of the command center and a view of the crew station (Fig. 20) where the Remote Manipulator System (RMS) and the Telerobotic Work Station (TWS) will be located. The command center is located in the bow of the ship and is where the commander (pilot), the navigator, and the payload specialist will be

stationed during flight. This command/control center (Fig. 21) consists of all the avionics equipment needed for control of the MOTV including computers for data processing, and displays and keyboards mounted on the command control panel. Cathode-ray tubes (CRT's) and light emitting diode (LED) displays will be used in conjunction with a keyboard consisting of buttons and toggle switches. The command center will also contain windows for visual capabilities.

Another compartment will contain more computer equipment, and the EVA equipment. These facilities placed throughout the cabin will be strategically positioned to ensure proper weight distribution. The vertical sleeper, waste collector and storage facilities will be located on one side of the cabin and the food gallery, personal hygiene center, and the controls for the ARS on the other. The crew station control panels (Fig. 22) for the RMS and the TWS as well as the airlock (Fig. 23) will be located on the back wall along with two small windows which provide the payload specialist true visual control capabilities.

Extra-vehicular Activity (EVA) Support - Extra-vehicular activities are required when a crew member must go outside the safety of the cabin environment to complete a task such as repairing, retrieving, or maintenance of a satellite. To protect the crew member from the space environment, a special suit will be worn. This suit is the Environmental Mobility Unit (EMU) (Fig. 24) and is a work of art within itself. This suit is a liquid cooled, pressurized, integrated thermal micrometeoroid garment that keeps the crewmember in a 100% oxygen environment (Ref. 27). The EMU consists of three assemblies: the upper torso, the lower torso, and the portable life support system (PLSS) (Fig. 24). Before donning the EMU, the crewmember must wear a ventilated undergarment (Fig. 25) which keeps the man cool throughout the EVA mission. The PLSS contains the communication system necessary to link the crew member with the MOTV. Each EMU is rechargeable and has a power supply of approximately 7 hrs, which gives enough time for 6 hrs of EVA. For the satellite recovery missions, a crewmember will be fitted

with a Manned Maneuvering Unit (MMU) (Fig. 26).

On each EMU is mounted a microcomputer with LED displays which supplies a constant oxygen and battery power check. Like the computers on the YSTAR, the microcomputer will provide a warning and specifies of any corrective actions needed to be taken in case of system failure or emergency. The EMU will be stored in the airlock located at the rear of the command/control cabin.

VII. COMMUNICATION SYSTEM

The communication system provides the essential capabilities that any space vehicle must have. YSTAR will need four different types of communication links. These links and their respective bands are shown in Table 2. The capabilities of the communication system are voice, telemetry, command, TV, data (analog, digital), EVA, and inter-vehicular. Intervehicular command system (IVCS), consisting of a headset and a communication control box will be worn by each crew-member (Fig. 27). The communication control box can be used as an onboard intercom or an external communicator for the EVA and other OTVs and for communication with the space station (Ref. 10). The control box, which measures about 4 x 4 x 5 inches and weighs about 2 lbs, can be connected to one of several intercom boxes throughout the cabin. This enables each crew member the ability to move around without losing communication with the others. An option to the control box is a wireless microphone which attaches to the headset. YSTAR will maintain communication using both types of IVCS.

On board computers will transmit and receive all the analog and digital data. An intricate part of YSTAR's computer system will be the computer's ability to respond to the crew. This type of computer system is known as computer friendly and is also capable of regulating air flow, temperature, and ECLSS functions (Ref. 27). Antennas will be used to transfer information to and from the MOTV. The antennas will be a mixture of VHF dish, S-band (retractable, steerable), Ku-band dish (retractable), and UHF dish and will range from 3 to 4 feet in size. Antenna location will be determined by mission requirements and design restrictions.

Table 2 Communications Links

	Transmit Band	Receive Band	Type
Space Station - VSTAR	Ku	Ku	Voice, TV, Data
	S	S	Voice
VSTAR - EVA	K	K	Voice, TV
VSTAR - OTV	K	K	Voice
	Ku	Ku	Data
Inter - Vehicular	K	K	Voice

VIII. ELECTRICAL POWER

Introduction

The electrical power system for VSTAR has several basic requirements. The first is that it has to be a system that can be safely used by humans. That is, for example, if nuclear power is used, shielding would be required to protect the crew. The second requirement is that the system must be capable of producing 4.9 kW continuous power and 6.0 kW peak power. The system must also have a total life of at least 10 years, a mission duration life of 72 hours, be easy to shut down and start up, and be as lightweight as possible.

The Power System Domain graph, figure 28, gives the basic guidelines for choosing a specific type of power system for a given mission duration time and required power output. VSTAR is shown to be best suited for fuel cells and solar arrays. New developments in the energy density and life of primary batteries, specifically lithium-thionyl chloride batteries, as shown in Vol. II, also make them a candidate for VSTAR.

Power Systems

In designing the electrical system for VSTAR, the following five power producing systems were studied:

1. Solar arrays
2. Radioisotope thermoelectric generators
3. Lithium-thionyl chloride batteries
4. Hydrogen-Oxygen fuel cells
5. Multi-fuel fuel cells

Solar Arrays - Solar arrays are basically used to convert the sun's energy to electrical energy by photovoltaic conversion. The array itself would extend several feet beyond the body.

The basic reason that solar arrays were not chosen for VSTAR is that maneuverability of the vehicle would be reduced. VSTAR has to be able to capture and repair satellites as well as dock at the space station. Large solar arrays would only prove to be a hinderance in these operations with a high possibility of damage. The concept of a mechanism that would repeatedly fold and unfold the array was considered but rejected. Besides the added weight of the mechanical system, repeated foldings would considerably increase the chances for array failure.

Radioisotope Thermoelectric Generators - Radioisotope thermoelectric generators (RTG's) are nuclear devices that convert the heat produced by the decay of a radioactive material, such as plutonium-238, to electrical power. RTG's were originally thought to be an option for VSTAR but the power requirements for VSTAR are much higher than first estimated. VSTAR has a 6 kW peak power requirement. The General Purpose Heat Source (GPHS) RTG, designed by General Electric, produces only 250 watts, weighs 122 lbs and has an estimated cost of \$1800/Watt. Based on these GPHS values an RTG the size VSTAR would require would weigh 2928 lbs and cost \$10.8 million, excluding the cost and weight for the required shielding (Ref. 29). Clearly the RTG is better suited for unmanned, low power requirement applications.

Hydrogen-Oxygen Fuel Cells - A H_2-O_2 fuel cell is a device that directly converts chemical energy to electricity. Figure 29 shows the schematics of this type of fuel cell. It is seen that the hydrogen and oxygen react with the potassium hydroxide solution setting up an electric potential with the reaction product being drinkable water that can be used by the VSTAR crew. One H_2-O_2 fuel cell of the size used on the Space Shuttle produces 7 kW of continuous power, weighs 202 lbs and is 14 x 17 x 40 inches (Ref. 12). For redundancy, VSTAR would require two such fuel cells. The hydrogen and oxygen would be stored in Dewar-type spherical tanks (Fig. 30 & 31) with a tank weight of 440 lbs (Ref. 26). The hydrogen tanks would have a

diameter of 45.5 inches and the oxygen tanks an diameter of 36.8 inches. The total H_2-O_2 fuel cell system would weigh almost 1300 lbs.

Lithium-Thionyl Chloride Batteries - The lithium batteries also directly convert chemical energy to electrical power. Jet Propulsion Laboratory, a company developing the battery, estimates that it will have an energy density of 250 Watt hr/lb and an active storage life of 5-10 years (Ref. 13). This is a great improvement, over the second best, silver-zinc batteries, in weight savings, life and cost. The system required for VSTAR, including redundancy, would require 10 such batteries at a total weight of 400 lbs. Each battery is 9.7 x 11.7 x 5.2 inches giving a total system volume of only 3.5 cubic feet (Ref. 13).

Multi-Fuel Fuel Cells - The MFFC's work on the same principal as the H_2-O_2 fuel cells, but can use any hydrocarbon fuel such as gasoline, alcohol, methane, jet fuel or gasified coal. The system is extremely small and light weight with thin layers between each alternating air and fuel passageway made of ceramic materials (Fig. 32). The air and fuel react electrochemically across these layers producing a current at a temperature of 800 - 1000 °C. A 15 x 15 inch cell is estimated to be able to produce 50 kW of power (Ref. 3), so that a small 3 x 3 inch cell would be capable of powering VSTAR. Oxygen and fuel tanks would be required as with the H_2-O_2 fuel cells, but the size of the tanks and amount of fuel has yet to be determined. The MFFC is expected to be available in about 10 years.

Each of the above systems are good VSTAR candidates. The MFFC's were chosen over the other two for several reasons. The MFFC's weigh less than the H_2-O_2 fuel cells, consequently the H_2-O_2 fuel cells were rejected. The MFFC's were chosen over the lithium batteries because

they have a longer expected life than the batteries. Although the batteries will most likely weigh less than the MFFC system, the MFFC's have a much higher working temperature. The extra heat produced can be used for VSTAR's environmental control therefore eliminating the extra weight of a separate heating system as required with the batteries. This would in effect reduce the overall weight of the MFFC system to less than the lithium battery system.

IX. DOCKING

Satellite Docking

VSTAR will be flown to within fifty feet of the powered down and passive spacecraft (satellite) where an astronaut in a space environmental suit will approach the satellite using a manned maneuvering unit (MMU) as shown in Figure. 34. The capture of the spacecraft will be similar to the Shuttle missions (Ref. 8) in that it will be accomplished using a mechanical assembly called a stinger, which is mounted on the MMU in front of the astronaut. The stinger is equipped with a long probe that will be inserted into the nozzle of spent apogee motor. The end of the probe contains three toggle lugs which will release when the probe is inserted. A jackscrew extending through the probe can then stabilize the satellite by forming a rigid connection between the spacecraft, stinger and MMU while the MMU control system and attitude thrusters despin and attitude stabilize the entire assemblage (Fig. 35). The astronaut will then maneuver the spacecraft back to the MOTV where it will be loaded into the payload bay. A telerobotic arm, like that of the Remote Manipulator System (RMS) used on the Space Shuttle, will be used to grapple the spacecraft at the stinger and secure it onto the supports of the truss structure.

This method of spacecraft capture has provided a safe and efficient means of docking with satellites in past shuttle missions. It is therefore perceived that this success will continue in the satellite recovery missions of VSTAR. There is an additional consideration however, that must be accounted for in the future SLR missions of VSTAR. Since a direct launch into GEO will be possible, future spacecraft will only require a small reaction control system for minor orbit corrections, rather than a massive solid rocket booster presently used to deliver satellites from Shuttle to geosynchronous orbit. This will mean that the docking procedures outlined above will require updating for application to future satellite designs. Through this might be accomplished in several ways, the most logical solution would likely be a simple modification of the stinger

control device. Since future spacecraft will not contain an apogee kick motor, the stinger could be equipped with an adaptor to despin and stabilize the spacecraft using a linkage device.

A new conceptual design called the telerobotic work station (TWS) will also be incorporated into YSTAR'S design to assist the astronauts in various repairing tasks. The TWS is designed by Martin Marietta Aerospace and consist of the robot work station and control station (Fig. 35). The robot work station consist of three dextrous arms, controlled from a remote control station on board the MOTV. Its capabilities include observational sensing, force sensing, tactical sensing, gripping and the use of tools. The TWS will work in conjunction with the RMS to reduce the crew time spent in hazardous operations, such as satellite refueling.

Space Station Docking

Docking to a space station will be accomplished by bringing the MOTV into coplaner orbit but at a slightly lower altitude than the space station. It will have a period slightly shorter than the space station and will eventually overtake it. The MOTV will then be maneuvered to within a few meters of the space station docking platform so that the orbital speeds are equal. The remaining distance will be closed through the use of mechanical grapplers; which will minimize changes in the original momentum of the space station and lessen any tendency for it to be "bumped" into an undesirable orbit. Once docked, movement of the MOTV will be restrained by a rigid support system.

X. MANAGEMENT

The scheduling and management of a design project is a complicated task. The development of VSTAR will demand extensive research and testing of the spacecraft components and the complete vehicle assembly. The development schedule showing estimated time for each phase is displayed in the VSTAR development timetable (Fig. 36).

The preliminary design phase will outline the basic design criteria needed to accomplish mission goals. A final proposal including the design criteria and projected costs of the primary spacecraft systems will be completed during this time frame.

In the second phase a more in-depth research and feasibility analysis will determine if the spacecraft is both economically and technically feasible. Research emphasis will be placed on the TWS and advanced OTV engine. Special consideration will be given to the development of a high pressure dual expander engine with variable throat nozzle, along with other systems that are essential in the achievement of mission objectives at lowest possible cost.

During the latter part of the research phase, component construction and testing will begin. Each individual module will be tested as separate units first, then the entire vehicle will be assembled for further testing. These tests will check the linkage systems for possible module interface or vehicle docking problems. During this phase checks of the electrical power system, propulsion units, flight and environmental controls, onboard computers, robotics, and other systems will be thoroughly examined. Environmental and guidance control systems, as well as the propulsion units will also undergo extensive testing. VSTAR will then be prepared for transportation into space.

Once ready all VSTAR components will be simultaneously delivered to low earth orbit and stored at the space station. After components have made the transfer to the space station, VSTAR will be ready for reassembly in orbit and will then undergo extensive operational mission testing.

The purpose of the operational mission tests is to decide if the MOTV is actually capable of executing the mission objectives. The mission complexity will range from simple maneuvering operations to the highly complicated tasks associated with SLR missions. Once VSTAR has proven itself worthy in the operational tests, it will then be available for commercial service. If all projected deadlines are met, VSTAR will be fully operational by the year 2001.

XI. COST ANALYSIS

To determine the economic feasibility of any project a cost analysis or prediction is required. By applying the NASA Space Station Cost Estimating Relationships (CER), a reasonable cost estimate can be made for the VSTAR project (Ref. 23). The results of this estimate is shown in Table 3. These results are based on a use frequency of 10 missions per year and are reflections of current STS costs as well as regular VSTAR overhaul requirements.

The cost analysis is divided into three sections: vehicle costs, management costs, and annual operating costs. Each section is further broken down into development and testing costs (D&T) and mission costs (MC). Vehicle costs are a compilation of the structure, electrical power, and propulsion system costs. Program management costs represent the money that must go into the organization of personnel as well as time invested in the VSTAR project. The initial assembly of VSTAR in orbit is considered as an addition to the mission cost of the vehicle. Operational costs are estimated without the use of the CER since this criteria does not apply to operations.

Averaging all costs over the 10 year life span of the vehicle, the cost per year of service is \$899.17 million. Each additional vehicle will add \$ 169.70 million in hardware and management cost. However, the average cost per pound of payload will decrease for the overall program. By the end of its design life each VSTAR will have transported 500,000 lbs of cargo to and from geosynchronous orbit. The above figures taken together result in a dollar per pound cost of \$1798.34. It is significant to recognize that 89.4% of this cost is coming from the cost of shuttle flights needed to lift VSTAR's fuel into orbit. Any reduction in the cost per pound to deliver fuel from earth to space will substantially lower the cost of VSTAR.

TABLE 3 • VSTAR COST SUMMARY (Millions of Dollars)

<u>Vehicle Cost</u>	<u>Wt</u>	<u>D&I</u>	<u>MC</u>
Structure	3200 kg	\$ 194.40	\$ 30.90
Electrical Power	220 kg	\$ 28.00	\$ 5.80
Propulsion	450 kg	\$ 45.40	\$ 6.70
Initial Construction		—	\$ 74.00
Total		\$ 267.80	\$ 117.40
<u>Management Costs</u>			
Integration, Assembly, Checkout		\$ 89.50	\$ 23.00
System Test and Evaluation		\$ 165.10	—
Systems Engineering and Integration		\$ 108.60	\$ 9.40
Program Management		\$ 75.30	\$ 9.90
Total		\$ 438.50	\$ 42.30
<u>Annual Operating Costs</u>			
Fuel (Including 11 STS Flights)			\$ 803.50
Addition/Replacement of Hardware			\$ 5.00
Program Management			\$ 44.00
General Operating Expense			\$ 2.50
Total			\$ 855.00/yr/vehicle

XII. OPTIMIZATION

The purpose of optimization is to design a vehicle that will meet the proposed mission requirements and do so in an economical manner. The main cost in the operation of the proposed orbital transfer vehicle is the transportation of the fuel to orbit. With this in mind, an "optimum" configuration would be one which performs the mission with a minimum amount of fuel expended. The problem dealt with here is how to design the optimum configuration. To get an idea of the complexity of the problem, consider the following:

Any increase in the burnout mass will increase the mass of fuel required to obtain the required Δv and any increase in the mass of fuel will increase the size of the fuel tanks and thus further increase the burnout mass.

Any increase in the burnout mass will decrease the maximum acceleration and allow for lighter supporting structures. This will in turn decrease the burnout mass and give higher accelerations.

A change in the maximum allowable pressure in the fuel tank will change:

1. The vapor pressure of the fuel, which will change the temperature, the heat of vaporization, and hence the mass evaporated.
2. The heat leak rate into the tank and hence the mass evaporated.
3. The stress in the tank walls, the mass of the tank, and hence the burnout mass.

There will be an evaporation loss and the size of the tank must be increased, but this then will increase the evaporation loss. An increase in the endurance will increase the mass of the life support system and the mass of evaporation from the tanks, thus increasing the burnout mass. A decrease in the tank insulation thickness will simultaneously decrease the burnout mass and increase the mass of fuel evaporated. As the acceleration increases, the pressure, density, and mass at the bottom of the tank increase, thus increasing the burnout mass.

The list could continue. To consider one component while ignoring of the interrelations among the other components would result in a highly inefficient structure, or worse, an unworkable configuration. The design of the components therefore must be considered simultaneously. A computer program has been written which calculates these parameters using iterative techniques.

It has been found convenient to divide the parameters describing the vehicle into three categories as shown in Table 4. They will be referred to as the design constants, the design variables, and the calculated parameters. This corresponds to the parameters which are given, the parameters which are to be chosen in order to optimize the configuration, and the parameters which are to be calculated to describe the configuration.

Table 4. Parameter Divisions

<u>Design Constants</u>	<u>Design Variables</u>	<u>Calculated Parameters</u>
ΔV	ISP	a_{max}
m_{pay}	OFR	a_{min}
T_{end}	P_{fuel}	R
T_{out}	T_{insul}	m_{cabin}
k_{insul}	Thrust	$m_{life\ support}$
r_{insul}		$m_{cargo\ module}$
Q_{max}		$m_{support\ structure}$
$r_{material}$		etc.

The design constants are known parameters which are dictated by mission requirements or chosen by what will optimize the configuration. The first three in the column are ΔV , m_{pay} , and T_{end} . The SLR mission requires a ΔV of approximately 29,000 ft/sec, 5000 lbm payload and an endurance of 72 hours. These values are the givens in the problem. The next values in the column, k_{insul} and r_{insul} , are the insulation conductivity and density. For any given insulation, the one with the smaller r/k ratio will provide the greatest insulating ability for a given mass of insulation. Therefore, the one with the smaller r/k value will be the best choice, assuming other factors such as cost and suitability to the space environment are satisfied. Similarly r/σ , where r is the density of the material to be used in the structure, and σ is the maximum allowable stress in the material, determine the best choice for material, again assuming satisfactory performance in the particular operating environment. For example, cryogenic fuel tanks mandate the use of stainless steels, which have a higher r/σ than aluminum, but steel is the more suitable material because of its superior characteristics at lower temperatures. However, under given stress conditions, support rods in a structure would weigh less if made of aluminum, hence the aluminum would be the better choice and provide a more efficient structure. This is assuming that other factors such as sublimation, corrosion resistance, meteorite deterioration, etc. are satisfied by both materials. The last of the design constants, T_{out} , is the outside equilibrium temperature of the fuel tanks. Low temperatures are desirable in order to minimize evaporation losses.

In the next column of parameters, the design variables, it is not obvious which values will optimize the design. They are either subtle functions of each other or the calculated parameters are complicated functions of them. The approach taken here is to calculate the remaining parameters in column three for as many different combinations of design variables as practical. From this the most desirable configuration can be selected.

The values in column three can be determined once the design variables and design constants are known. As a group they are sole functions of the parameters in columns one and two, but individually they are complicated if not transcendental functions of the other parameters in

column three. In this case they are calculated by using iterative methods. The procedure used is summarized as follows:

1. Calculate the masses of the components of the vehicle and sum them, iterating where necessary.
2. Find the minimum and maximum accelerations from the sum of the component masses and reiterate.
3. Find the mass ratio needed from ΔV and ISP.
4. Find the mass of fuel required from the sum of the masses and the mass ratio and reiterate.

The result of this is a structure that is completely "balanced" for the particular set of initial conditions, i.e.; the mass of the structure is exactly the mass needed to withstand the maximum accelerations, the thickness of the tanks is exactly the thickness needed to withstand the pressure generated at the bottom of the tank due to acceleration effects and vapor pressure, the volume of the tanks is just the volume needed to contain the fuel used during burns plus the fuel expected to evaporate away, and so on. The structure is now optimized for that particular set of initial conditions.

For analysis purposes, the configuration is broken into the components (Fig. 37). This allows simple mathematical expressions to be written for the masses of the individual components. The following is an overview of the program flow. The masses of the individual components (Fig. 37) can be expressed in the following form :

$$m_{\text{cabin}} = f(m_{\text{crew}}, m_{\text{air}}, m_{\text{structure}}, m_{\text{shielding}}, m_{\text{other}})$$

$$m_{\text{life support}} = f(\# \text{ of crew}, T_{\text{end}})$$

$$m_{\text{cargo module}} = f(m_{\text{cabin}}, m_{\text{life support}}, A_{\text{max}})$$

$$m_{\text{tank}} = f(m_{\text{fuel}}, m_{\text{evap}}, p_{\text{fuel}}, r_{\text{fuel}}, a_{\text{min}}, a_{\text{max}}, r_{\text{metal}})$$

$$m_{\text{evap}} = f(h_{\text{vapor}}, k_{\text{insul}}, t_{\text{insul}}, T_{\text{end}}, T_{\text{fuel}}, T_{\text{out}}, A_{\text{tank}})$$

At this point the first iteration occurs. The value for m_{evap} is substituted into the expression for m_{tank} . When two consecutive values for m_{evap} are within a specified difference, the iterations stop and program flow continues :

$$m_{insul} = f (A_{tank}, t_{insul}, r_{insul})$$

$$m_{support\ structure} = f (m_{cabin}, m_{life\ support}, m_{tank}, m_{insul}, m_{fuel}, m_{evap}, a_{max})$$

$$m_{engine} = f (m_{pumps}, m_{turbine}, m_{nozzle})$$

$$m_{brnout} = f (m_{cabin}, m_{life\ support}, m_{cargo\ module}, m_{payload}, m_{tank}, m_{insul}, m_{evap}, m_{support\ structure}, m_{fuel})$$

$$m_{total} = f (m_{brnout}, m_{fuel})$$

$$a_{max} = f (m_{brnout}, thrust)$$

$$a_{min} = f (m_{total}, thrust)$$

At this point in the program the second iteration occurs. The value for a_{max} is substituted into the expression for $m_{cargo\ module}$. When two consecutive values of a_{max} are within a specified difference the iterations stop and program flow continues :

$$mass\ ratio = f (\Delta V, ISP)$$

$$m_{fuel} = f (m_{brnout}, mass\ ratio)$$

Once the mass ratio and m_{fuel} are calculated the final iteration occurs. The value for m_{fuel} is substituted into the expression for m_{tank} . When two consecutive values of m_{fuel} are within a specified difference the iterations stop and program flow continues. When any unknown

value is encountered, it is initially assumed zero. As the program continues execution, the initial assumption is replaced by a calculated one. Eventually the values converge and the program ends the looping.

Six configurations were examined using the computer program. The configurations analyzed include engines with ISP's from 420 to 492 seconds, oxidizer fuel ratios of 6 and 7, and three dual fuel systems with oxidizer secondary fuel ratios of 2.21 and 4.25. All other design variables were held constant. All of the configurations were run assuming an engine thrust of 30,000 lbf. This produced moderate accelerations and a slightly higher mass of fuel required than for configurations using a 15,000 lbf. thrust. However, the overall burn times were reduced exactly by half. It is believed that the minimal savings gained from the reduced thrust is negligible in comparison to the increased operating life due to the shorter burn times. It is also noted that the high accelerations at burnout due to the larger thrust can be minimized and the burn times still held at a minimum if the engine is assumed throttleable. This was not taken into account in the configurations examined. The output for the two best configurations examined is shown in the Tables (5 , 6).

From the results of the program runs it is seen that ISP is the single most important variable in optimizing the configuration. Regardless of the other design variables, the configuration with the highest ISP consistently yielded lower fuel masses required. This may seem obvious on casual inspection but is not always true. For a given ISP, the configuration with the higher oxidizer fuel ratio yielded the lower mass of fuel required, and the configurations employing the dual-fuel systems were even better. This can be accounted for by the fact that for a given m_{fuel} , the higher OFR's require less LH₂. Since the density of LH₂ is so low, requiring enormous fuel tanks, the higher OFR configurations allow much of the dead weight of a large tank to be eliminated. It is possible then for a low ISP high OFR configuration to be superior to a high ISP, low OFR configuration. However, this was not the case for any of the configurations examined.

Due to time limitations, no optimization was attempted on the thickness of the insulation required on the fuel tanks, nor on the pressure inside the fuel tanks. For the configurations run these were held a constant 0.25 inches and 20 psi for all tanks.

The program generated output shown in the following tables includes the specific numbers for the components and the mass of fuel required by a given configuration for the SLR mission. Following this is a detailed description of the fuel tanks and of the thermodynamic state of the fuel within the tank. Figure 38 is a graphic comparison of the six configurations analyzed. The mass of fuel required for the SLR mission is shown on the y-axis for each configuration. The different configurations are plotted on the x-axis. The magnitude of influence of increasing ISP's and the influence of the dual-fuel system for a given ISP is evident from the bar graph representation. The fuel chosen for VSTAR is the bipropellant with an ISP of 492. This gave the second lowest mass of fuel required of the configurations examined.

Table 5 - Optimization Program Output (Lox/H₂/RP-1)

OXYGEN/HYDROGEN RATIO	[.]	:	6
OXYGEN/RPJ RATIO	[.]	:	2.21
ISP	[.]	:	492
ACCELERATION - MAX.	[g's]	:	2.53
ACCELERATION - MIN.	[g's]	:	0.41
DELTA V - OBTAINABLE	[ft/sec]	:	28999.99
MASS RATIO	[.]	:	6.237193
STUCTUAL COEFF.	[.]	:	0.0995
PAYLOAD COEFFICIENT	[.]	:	0.0724
TIME, ENDURANCE	[hours]	:	72
THRUST	[lbf]	:	30000

MASSES OF COMPONENTS :

CABIN	[lbm]	:	2247
LIFE SUPPORT MODULE	[lbm]	:	1376.86
CARGO MODULE	[lbm]	:	520.5203
CARGO	[lbm]	:	5000
SUPPORT STRUCTURE	[lbm]	:	639.5573
HYDROGEN TANK	[lbm]	:	620.5325
OXYGEN TANK	[lbm]	:	257.7574
RPJ TANK	[lbm]	:	83.84286
ENGINE	[lbm]	:	1000
TOTAL BURNOUT MASS	[lbm]	:	11870.25
MASS OF FUEL	[lbm]	:	62166.8
TOTAL MASS	[lbm]	:	74037.05

		HYDROGEN	OXYGEN	RPJ
TANK - MASS	[lbm]	620.53	257.76	83.84
- MAX. STRESS	[lbf/in ²]:	35000.00	35000.00	35000.00
- METAL DNSTY	[lbm/ft ³]:	489.00	489.00	489.00
- PRESS., BOT.	[lbf/in ²]:	20.17	21.97	21.88
- RADIUS	[ft]	7.05	5.11	3.52
- SURFACE AREA	[ft ²]	624.57	328.44	155.77
- THICKNESS	[in]	0.0244	0.0193	0.0132
- VOLUME	[ft ³]	1467.73	559.69	182.81
- VOLUME, MTL	[ft ³]	1.27	0.53	0.17
PROP. - VAPOR PRS	[lbf/in ²]:	20.00	20.00	20.00
- DENSITY	[lbm/ft ³]:	4.38	68.64	95.11
- MASS	[lbm]	6399.10	38394.59	17373.12
- TEMP.	[rankine]:	38.38	166.41	58.33
- HEAT.VAPRZ	[btu/lbm]:	188.16	91.62	100.00
SURFACE TEMPERATUE	[rankine]:	600.00	600.00	600.00
INSULATION - RHO	[lbm/ft ² -in]:	0.20	0.20	0.20
- MASS	[lbm]	31.23	16.42	7.79
- THCKNS	[in]	0.25	0.25	0.25
-K	[btu-in/hr-ft ²]:	0.000056	0.000056	0.000056
HEAT LEAK - RATE	[btu/hour]:	78.57	31.90	18.90
- TOT	[btu]	5657.18	2296.76	1360.80
MASS EVAPORATED	[lbm]	30.07	25.07	13.61

Table 6 Optimization Program Output (LOH/LH₂)

OXYGEN/HYDROGEN RATIO [.] : 6
 ISP [.] : 492
 ACCELERATION - MAX. [g's] : 2.47
 ACCELERATION - MIN. [g's] : 0.40
 DELTA V - OBTAINABLE [ft/sec] : 29000
 MASS RATIO [.] : 6.237194
 STUCTUAL COEFF. [.] : 0.1011
 PAYLOAD COEFFICIENT [.] : 0.0706
 TIME, ENDURANCE [hours] : 72
 THRUST [lbf] : 30000

MASSSES OF COMPONENTS :

CABIN [lbm] : 2247
 LIFE SUPPORT MODULE [lbm] : 1376.86
 CARGO MODULE [lbm] : 511.7303
 CARGO [lbm] : 5000
 SUPPORT STRUCTURE [lbm] : 641.161
 HYDROGEN TANK [lbm] : 882.4154
 OXYGEN TANK [lbm] : 369.6116
 ENGINE [lbm] : 1000
 TOTAL BURNOUT MASS [lbm] : 12158.7
 MASS OF FUEL [lbm] : 63677.46
 TOTAL MASS [lbm] : 75836.15

		HYDROGEN	OXYGEN
TANK - MASS	[lbm] :	882.42	369.61
- MAX. STRESS	[lbf/in ²]:	35000.00	35000.00
- METAL DNSTY	[lbm/ft ³]:	489.00	489.00
- PRESS., BOT.	[lbf/in ²]:	20.19	22.17
- RADIUS	[ft] :	7.93	5.75
- SURFACE AREA	[ft ²] :	789.36	415.22
- THICKNESS	[in] :	0.0274	0.0218
- VOLUME	[ft ³] :	2085.40	795.59
- VOLUME, MTL	[ft ³] :	1.80	0.76
PROP. - VAPOR PRS	[lbf/in ²]:	20.00	20.00
- DENSITY	[lbm/ft ³]:	4.38	68.64
- MASS	[lbm] :	9096.78	54580.68
- TEMP.	[rankine] :	38.38	166.41
- HEAT. VAPRZ	[btu/lbm] :	188.16	91.62
SURFACE TEMPERATUE	[rankine] :	600.00	600.00
INSULATION - RHO	[lbm/ft ² -in]:	0.20	0.20
- MASS	[lbm] :	39.47	20.76
- THCKNS	[in] :	0.25	0.25
-K	[btu-in/hr-ft ²]:	0.000056	0.000056
HEAT LEAK - RATE	[btu/hour]:	99.30	40.33
- TOT	[btu] :	7149.83	2903.62
MASS EVAPORATED	[lbm] :	38.00	31.69

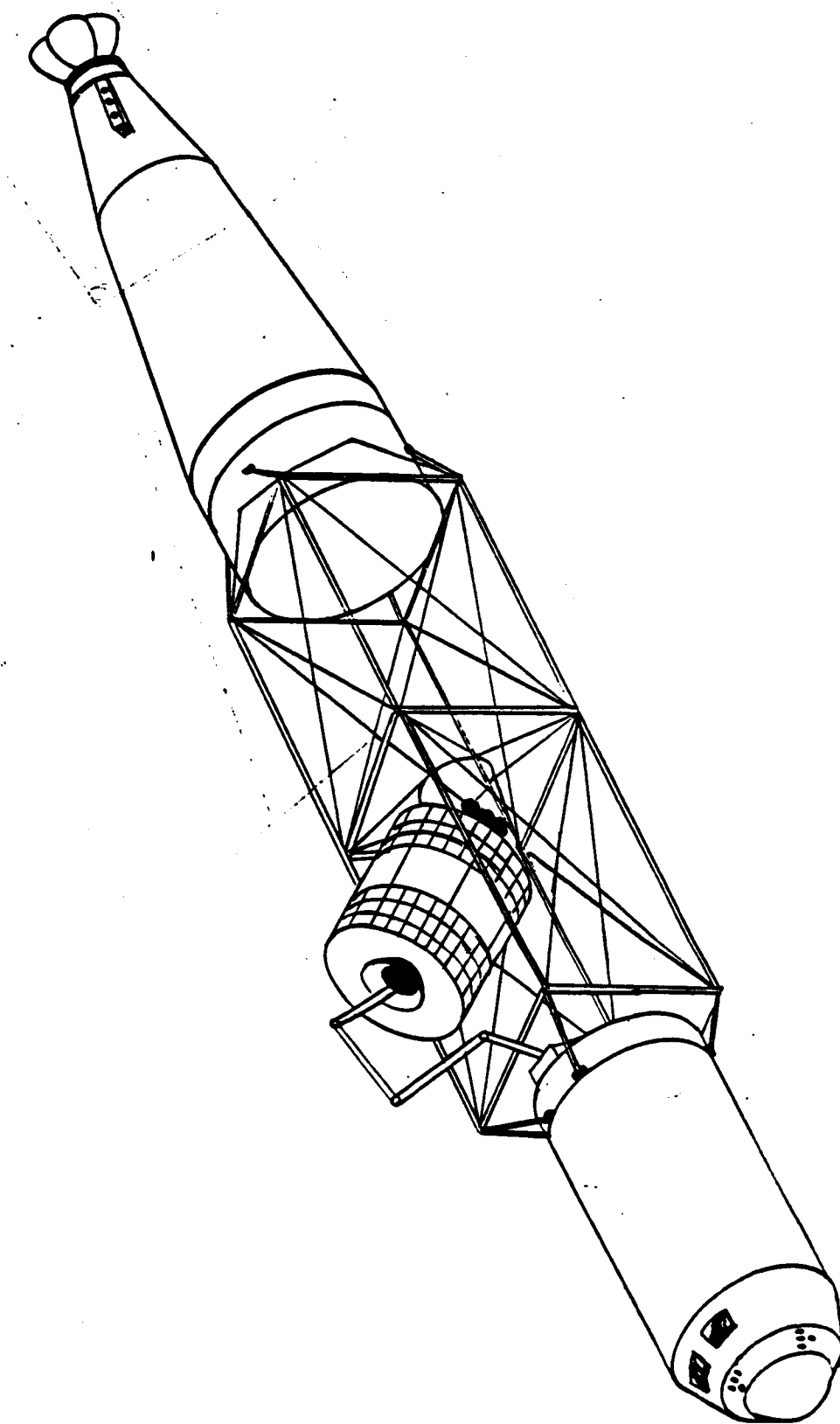


Figure 1 - USTAR Vehicle Configuration

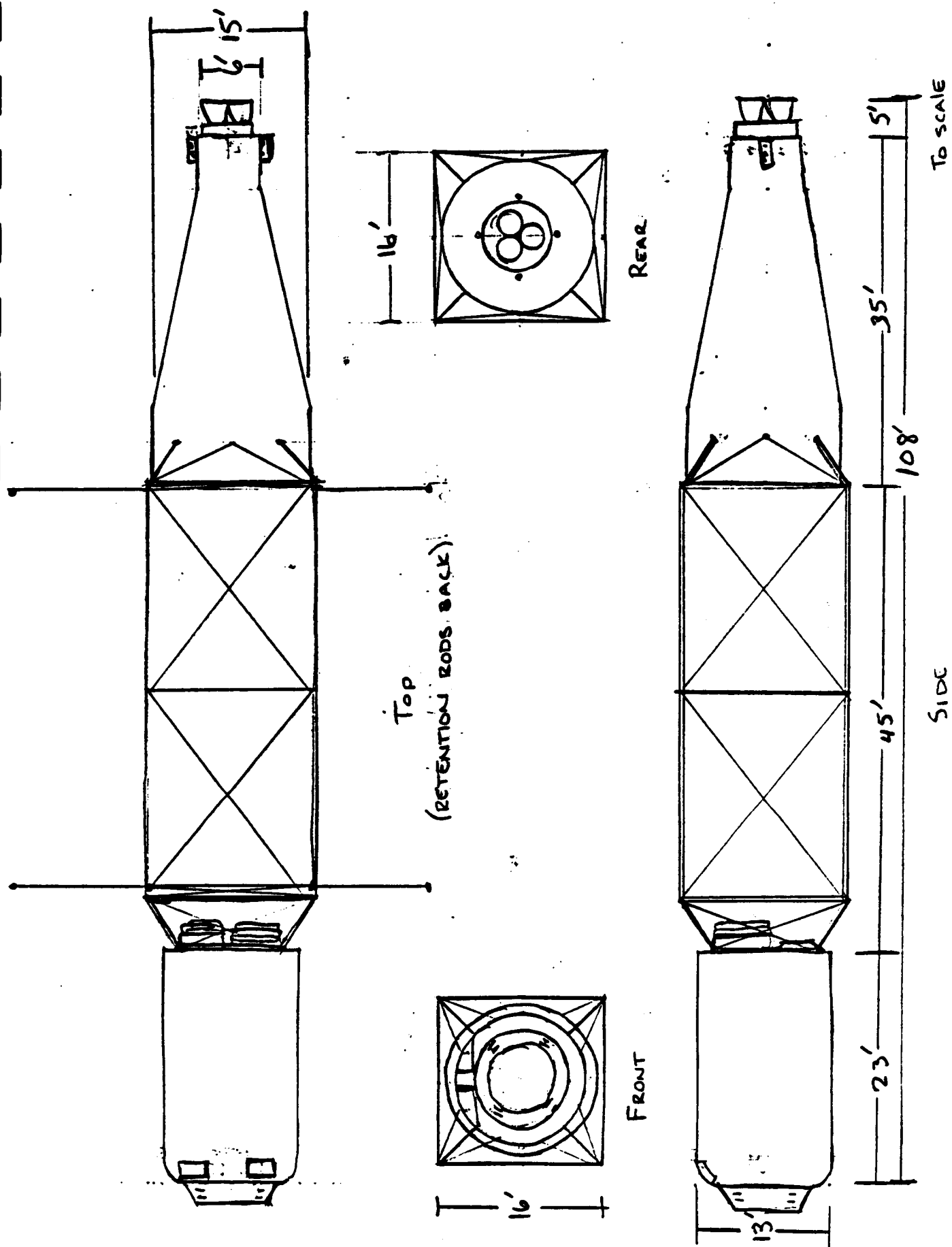
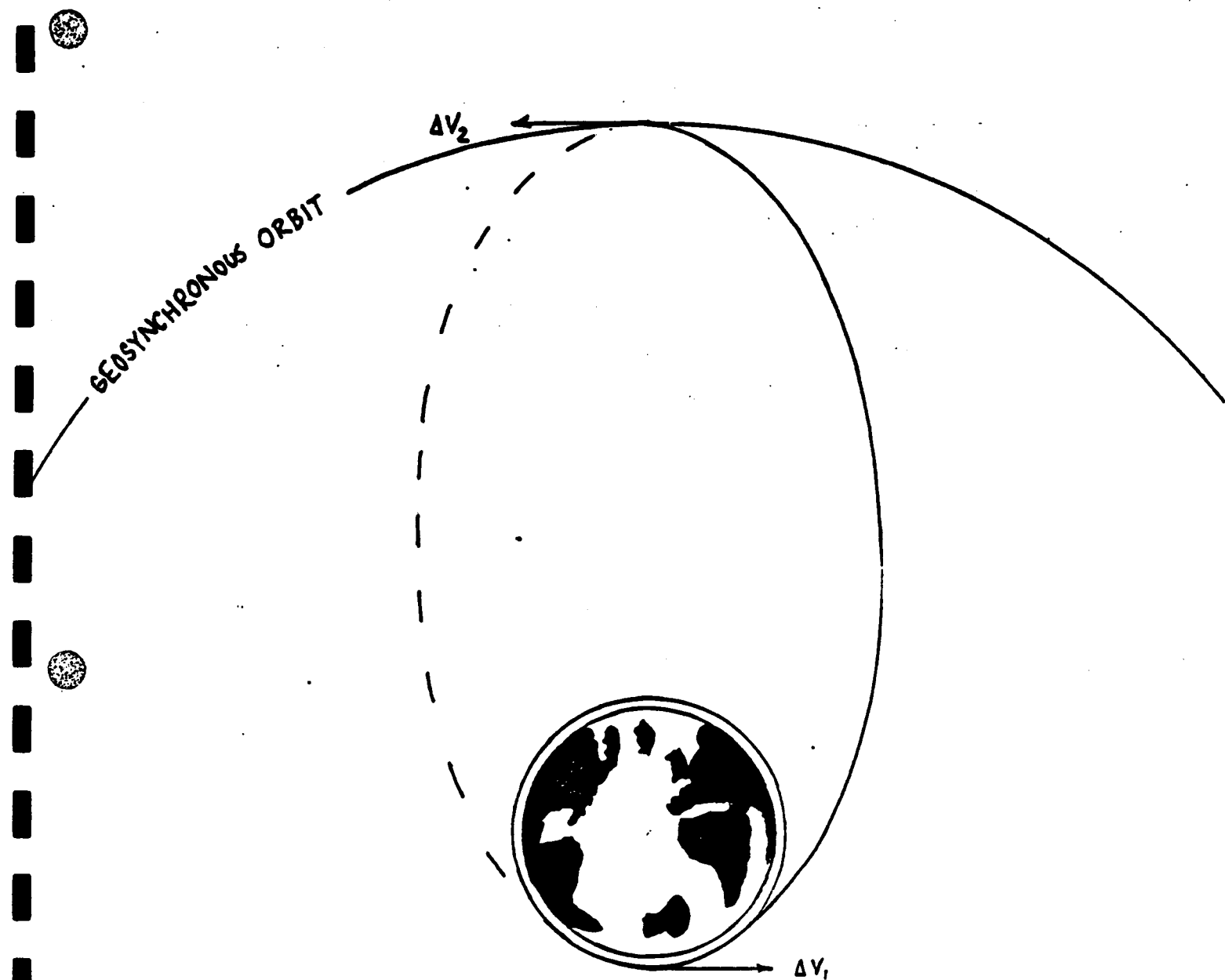


Figure 2 - USTAR Multiple Views



1 INCH = 5000 MILES

Figure 3 - Hohmann Coplanar Transfer

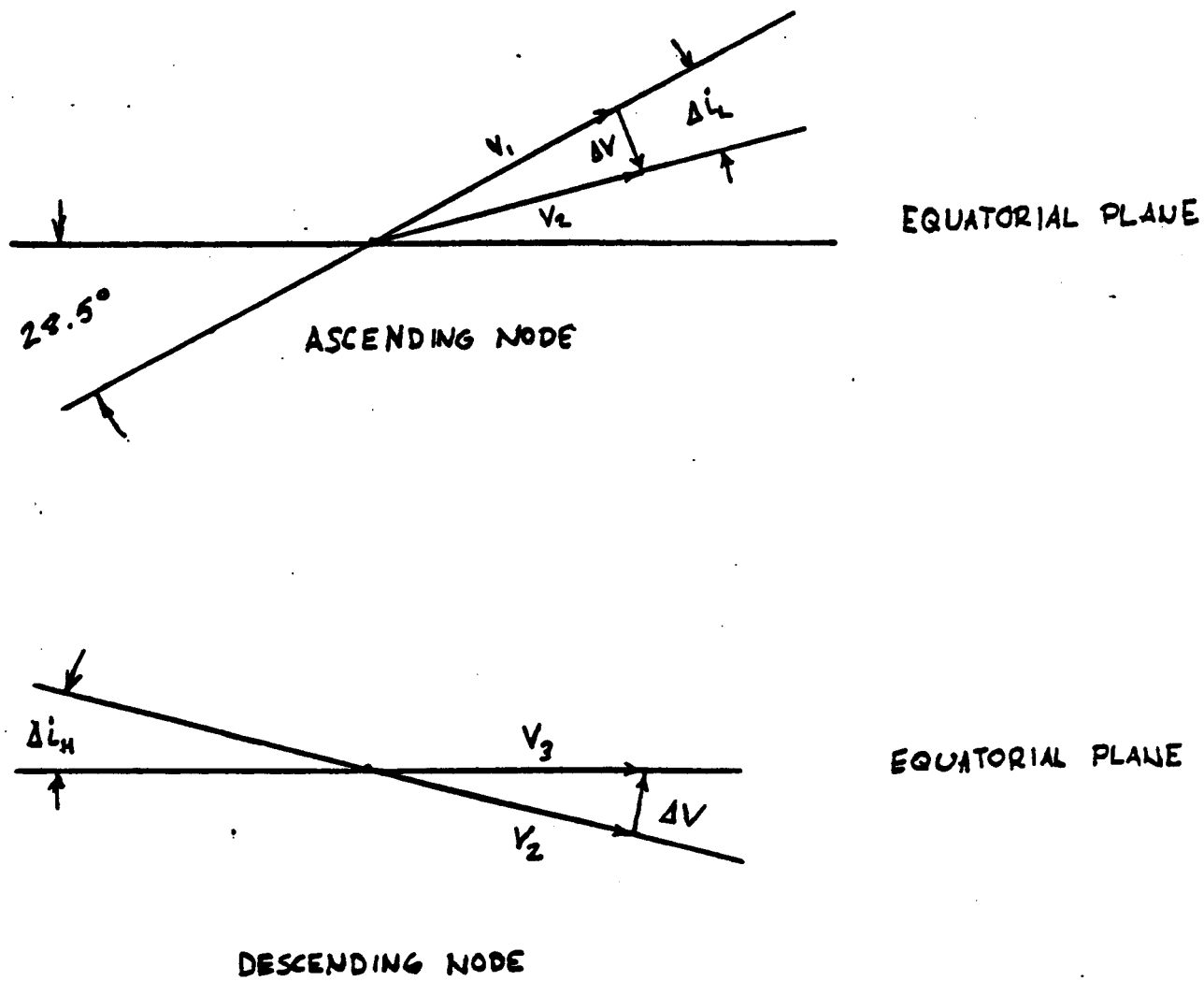


Figure 4 - Two-Impulse Maneuver Plane Change

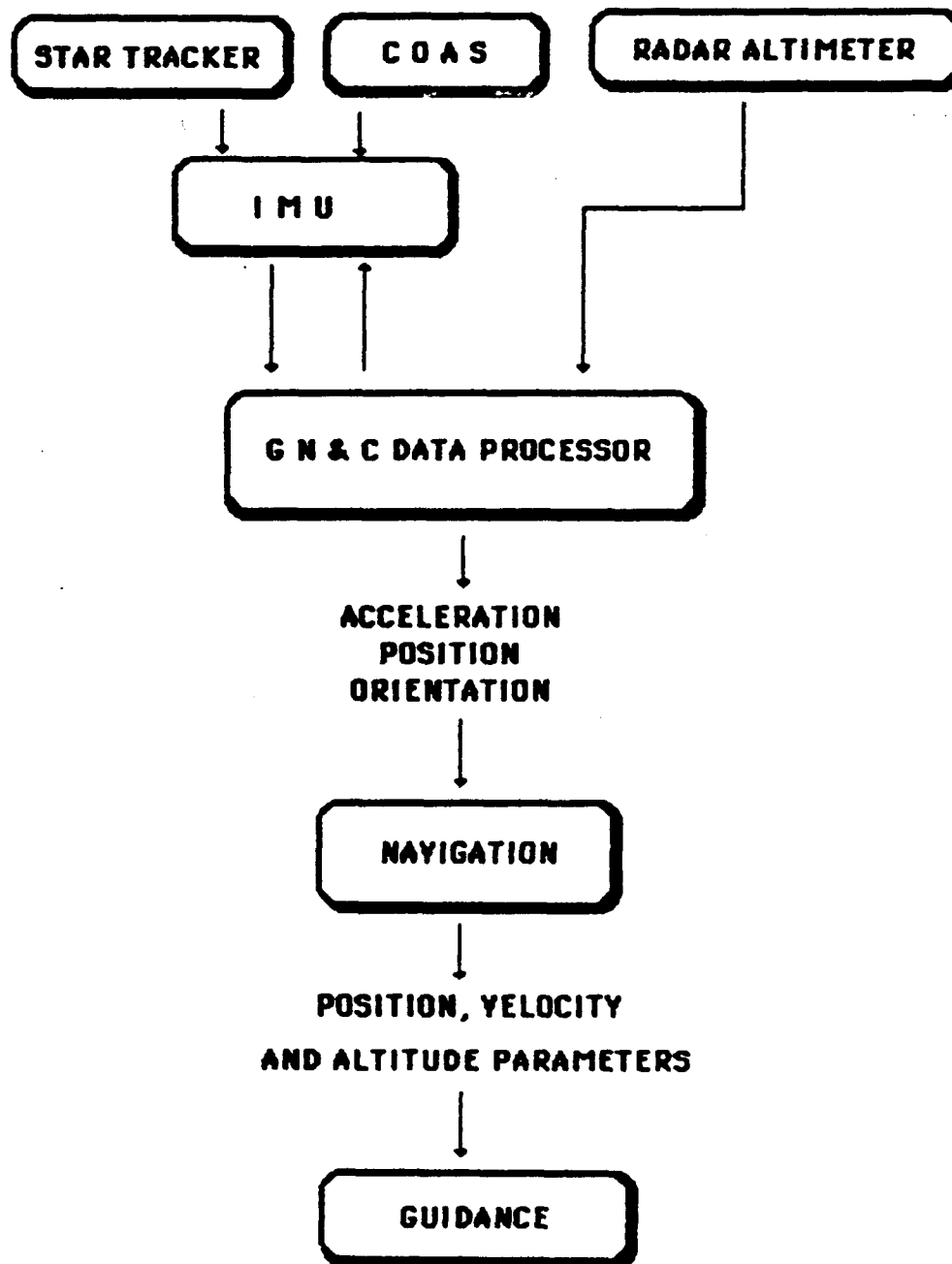


FIGURE 5 - G N & C SYSTEM SCHEMATIC (Ref. 29)

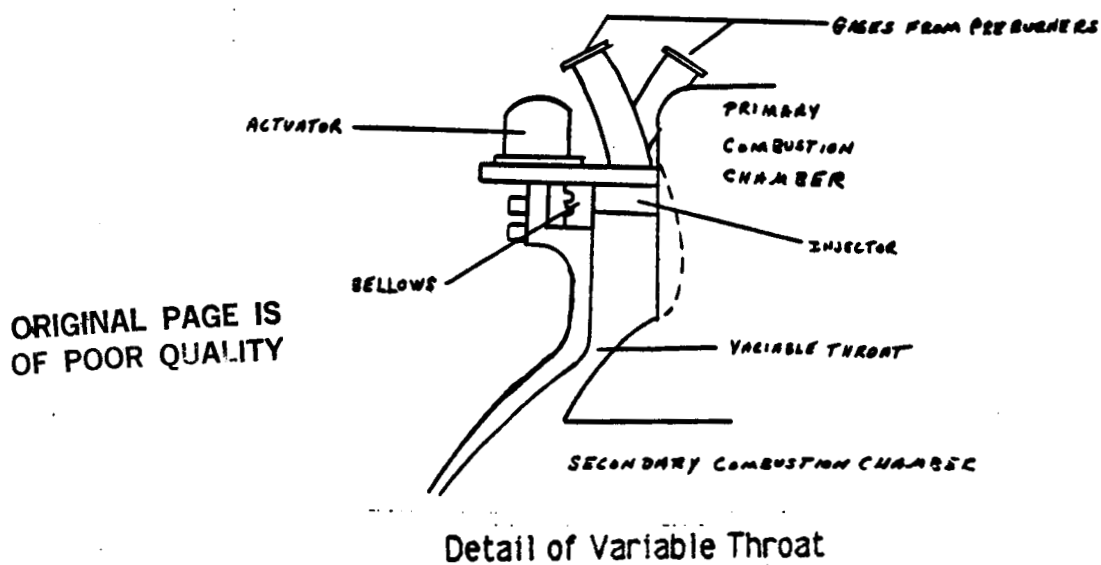
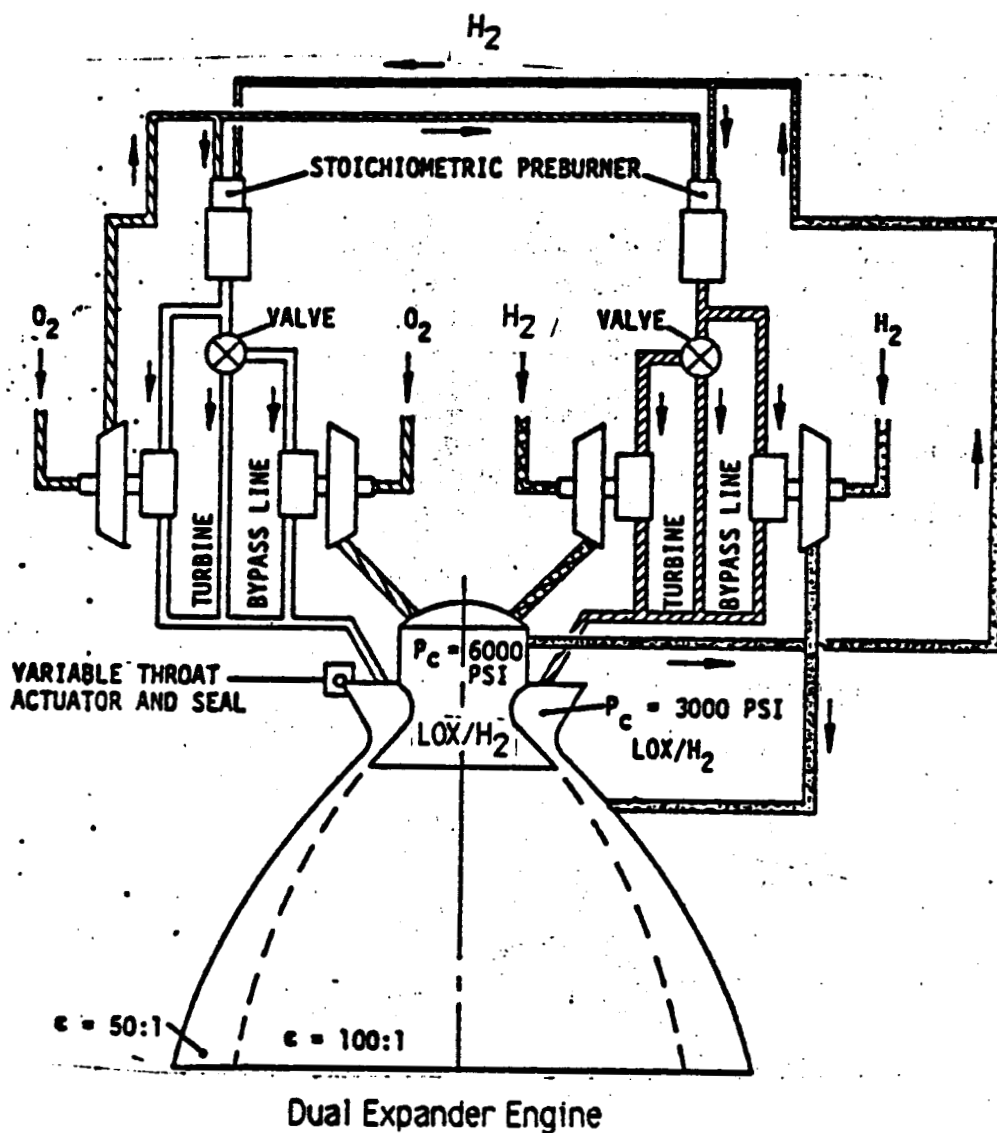
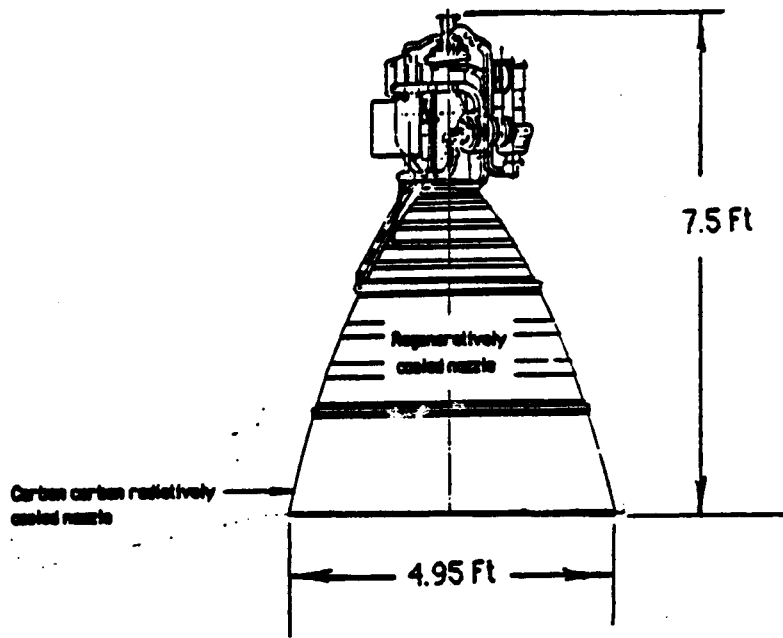


Figure 6 - Dual Expander Engine with Variable Throat (Ref. 5)

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SPECIFICATIONS

Isp:	492 secs
Propellant:	LOX/LH ₂
Mixture Ratio (Oxygen/Hydrogen):	6:1
Throttle Ratio:	30:1
Thrust Maximum:	15000 lbf
Thrust Minimum:	500 lbf

Figure 7 - Rocketdyne Advanced OTV Engine (Ref. 25)

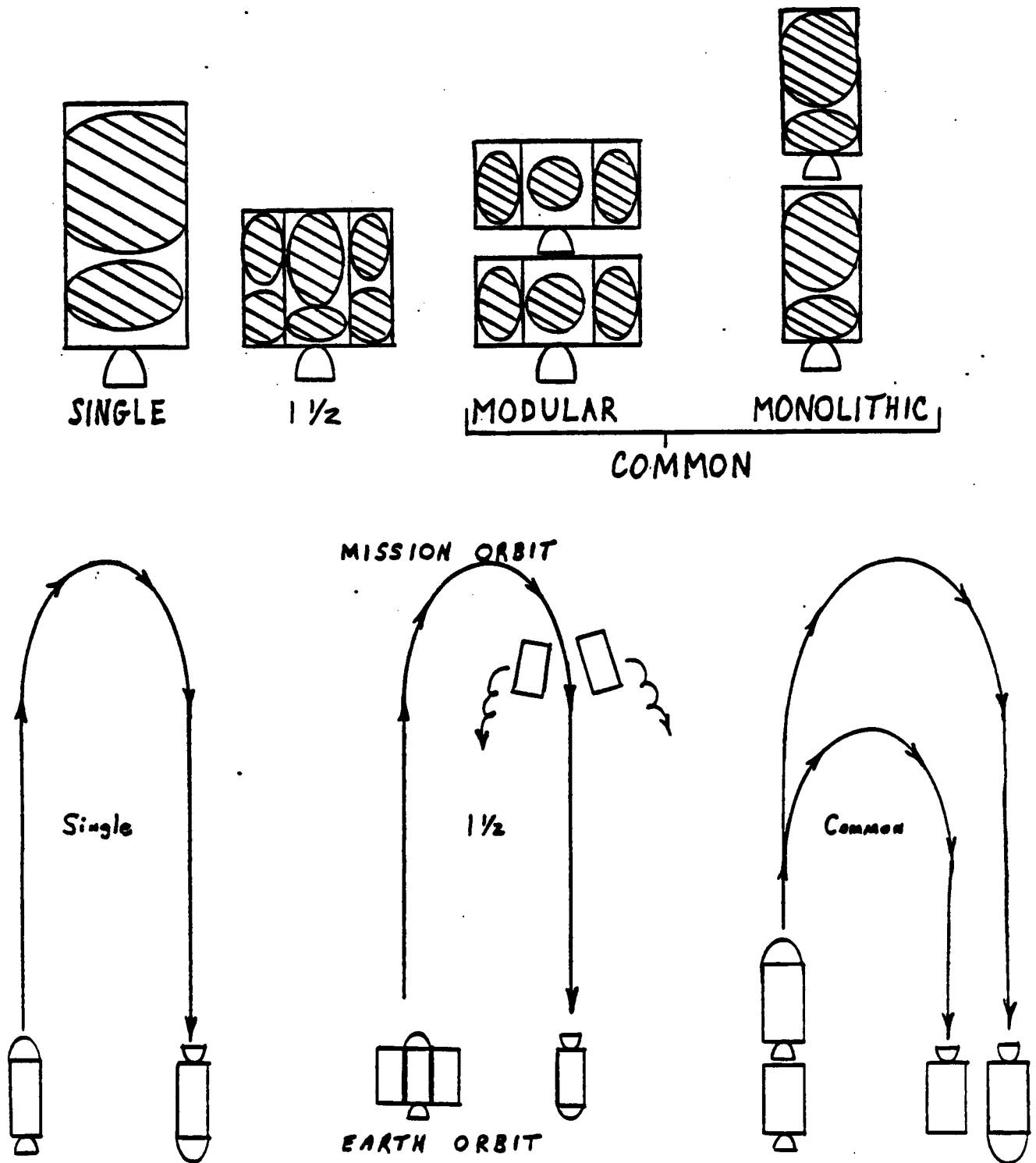


Figure 8 - MOTU Propulsion Configuration Candidates

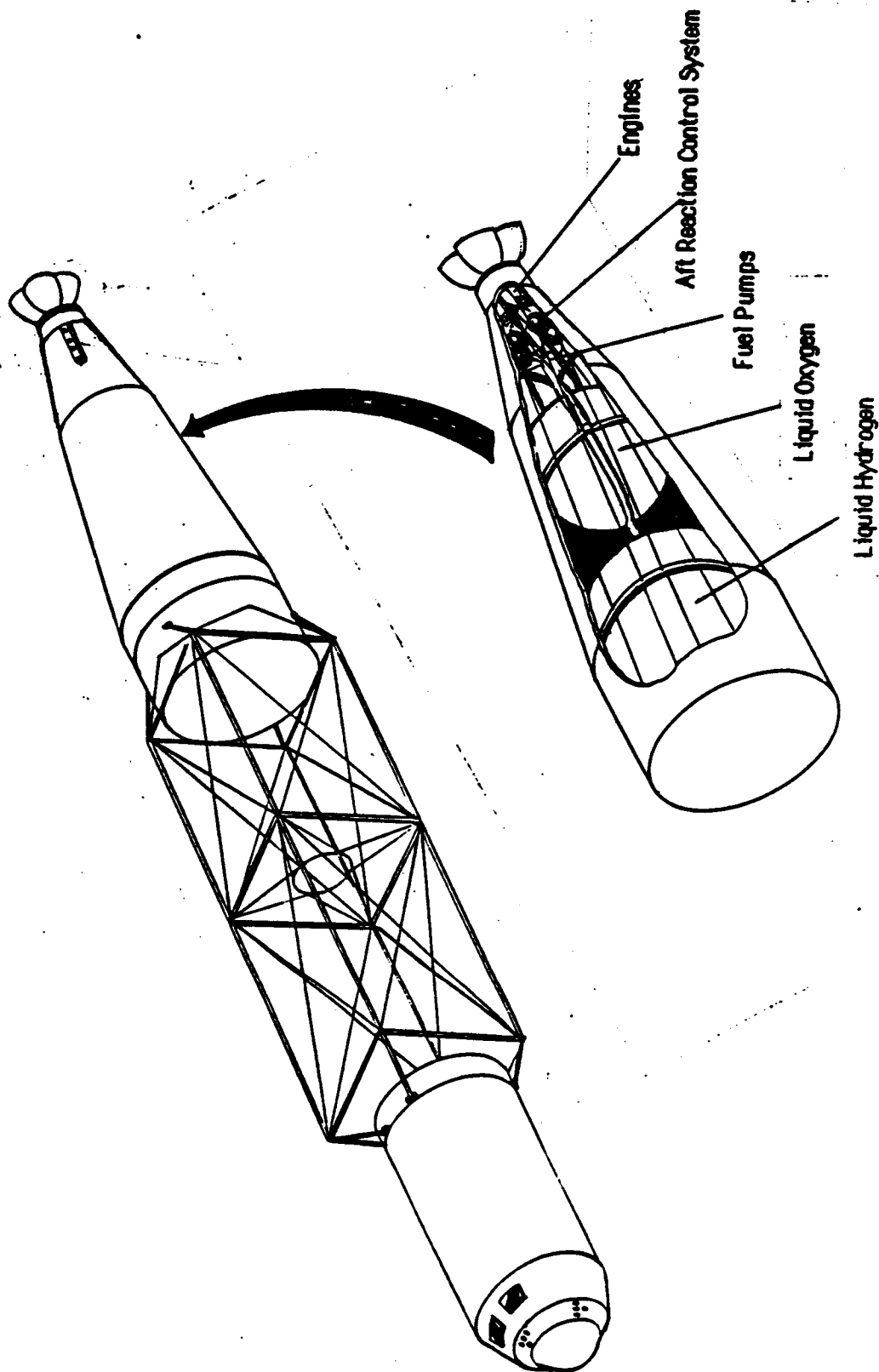


Figure 9 - Internal View of Propulsion Module

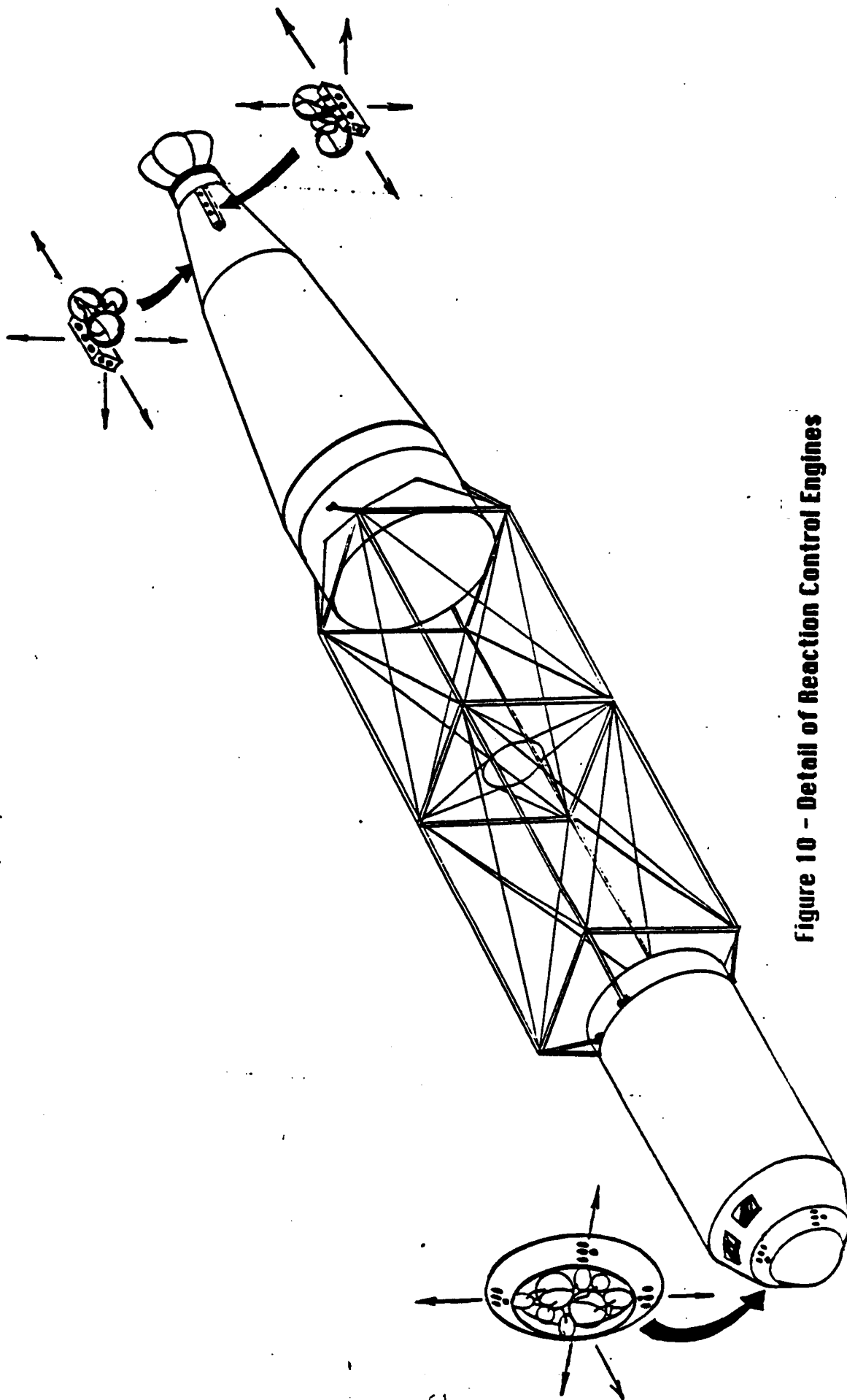


Figure 10 - Detail of Reaction Control Engines

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MARQUARDT R-40A

This precision control rocket was developed and qualified for the Space Shuttle Orbiter vehicle and is currently being used as an orbit insertion engine for the US Navy's Shuttle Launch Dispenser.

TYPE: Liquid propellant reaction control rocket.

PROPELLANTS: Nitrogen tetroxide and monomethyl hydrazine.

THRUST CHAMBER ASSEMBLY: Single chamber. Area ratio 20. Made of silicide coated columbium, with welded-on orthogonal and scarfed nozzle extension in same material. Internal film cooling. Exterior insulated for buried installation. Started by electrical signal to on/off solenoid valve. Multiple doublet injector with hypergolic ignition.

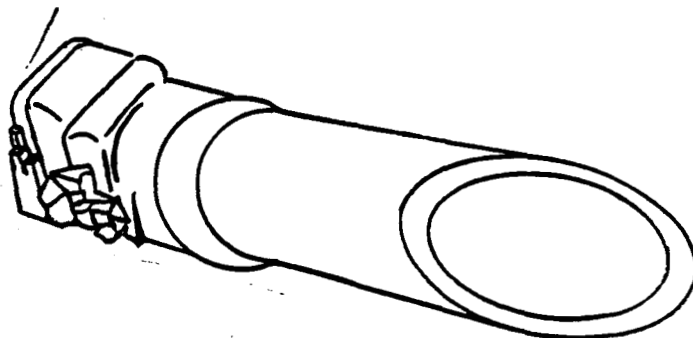
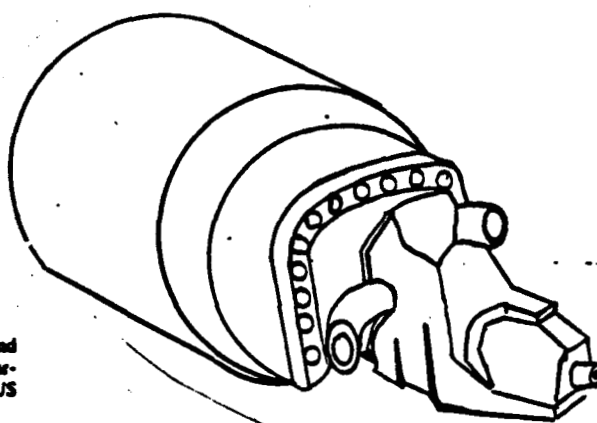
PROPELLANT FEED SYSTEM: Pressurized tanks, with feed of 0-526 kg (1-16 lb)/s fuel and 0-838 kg (1-85 lb)/s oxidant at 16-4 bars (238 lb/sq in abs).

DIMENSIONS:

Length overall	472 mm (18-6 in)
Nozzle exit diameter	267 mm (10-5 in)
WEIGHT, Dry:	9-5 kg (21-0 lb)

PERFORMANCE RATINGS:

Max thrust (vacuum)	3-87 kN (870 lb)
Chamber pressure	10-5 bars (152 lb/sq in)
Specific impulse: area ratio 20	281
area ratio 120	306



MARQUARDT R-1E

This small high performance rocket was qualified as the vernier engine for the Space Shuttle Orbiter.

TYPE: Liquid bipropellant rocket for use in space.

PROPELLANTS: Nitrogen tetroxide and monomethyl hydrazine.

THRUST CHAMBER: Single chamber. Minimum area ratio 26 with orthogonal and scarfed nozzles. Made of silicide coated columbium. Insulated for buried installation. Started by electrical signal to on/off solenoid valve. Single doublet injector with hypergolic ignition.

PROPELLANT FEED SYSTEM: Pressurized tank. Flow rate 0-016 kg (0-0354 lb)/s fuel and 0-0256 kg (0-565 lb)/s oxidant.

DIMENSIONS:

Length overall	279 mm (11-0 in)
Width	147 mm (5-8 in)
Height (depth)	145 mm (5-7 in)
WEIGHT, Dry:	3-7 kg (8-2 lb)

PERFORMANCE RATINGS:

Max thrust (vacuum)	0-11 kN (25-0 lb)
Chamber pressure	7-45 bars (108 lb/sq in)
Specific impulse (area ratio 100)	290

Figure 11 - USTAR Reaction Control Engines (Ref. 12)

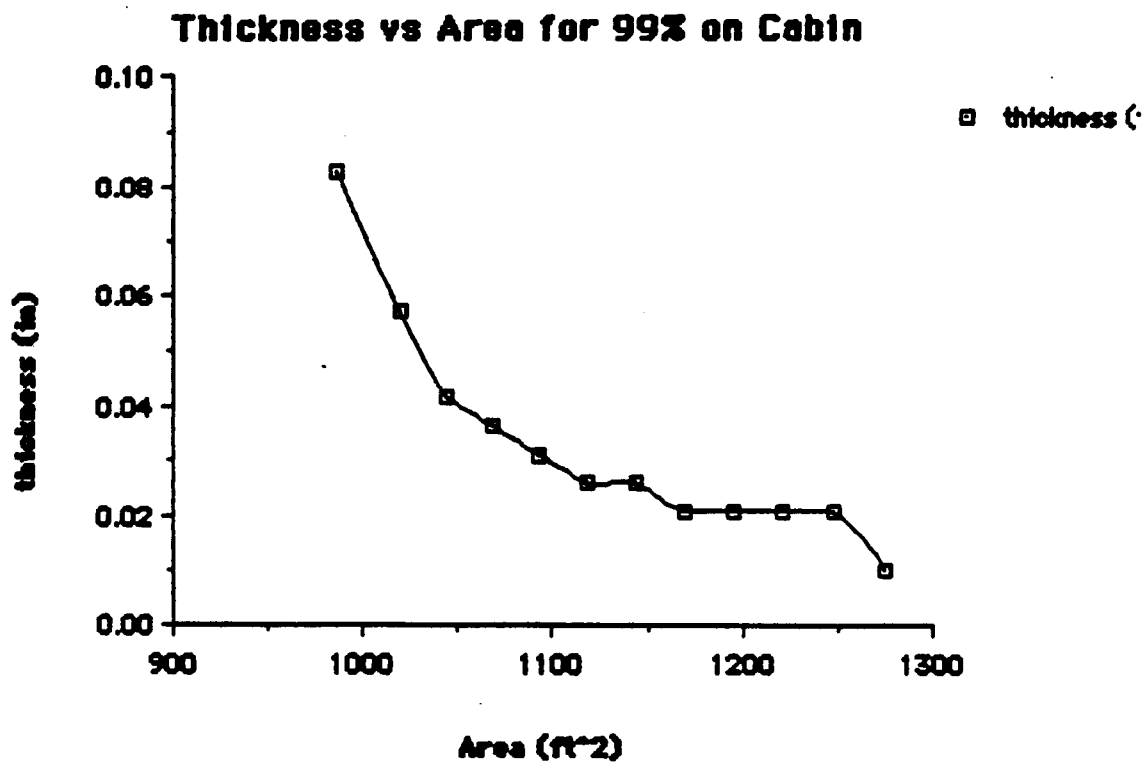
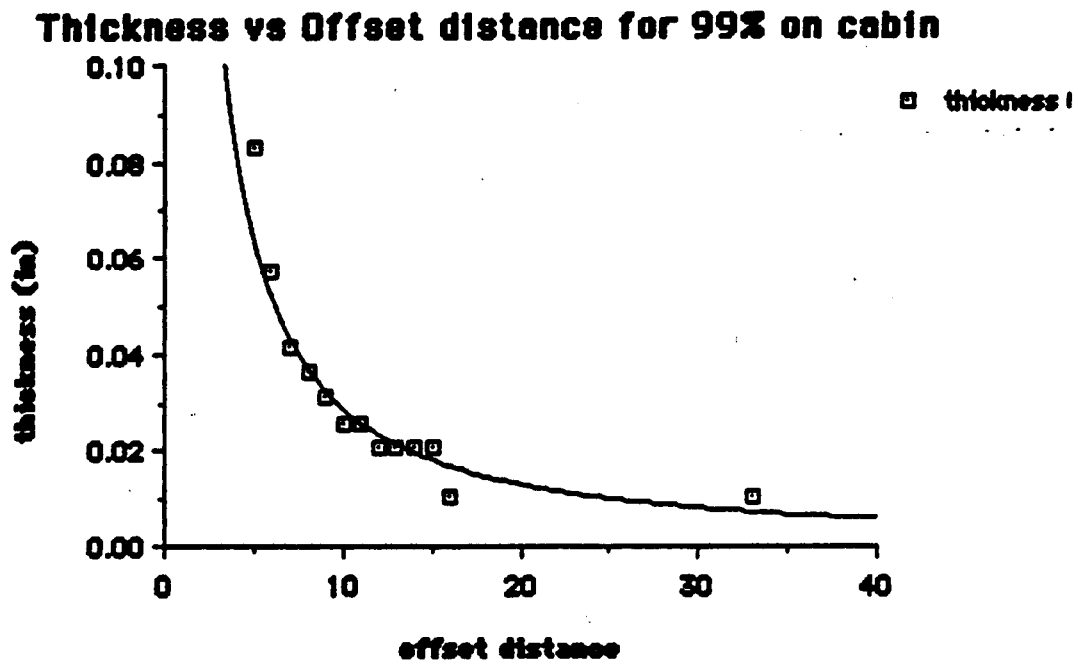


Figure 12 - Thickness Vs. Offset for Cabin Meteoroid Shield

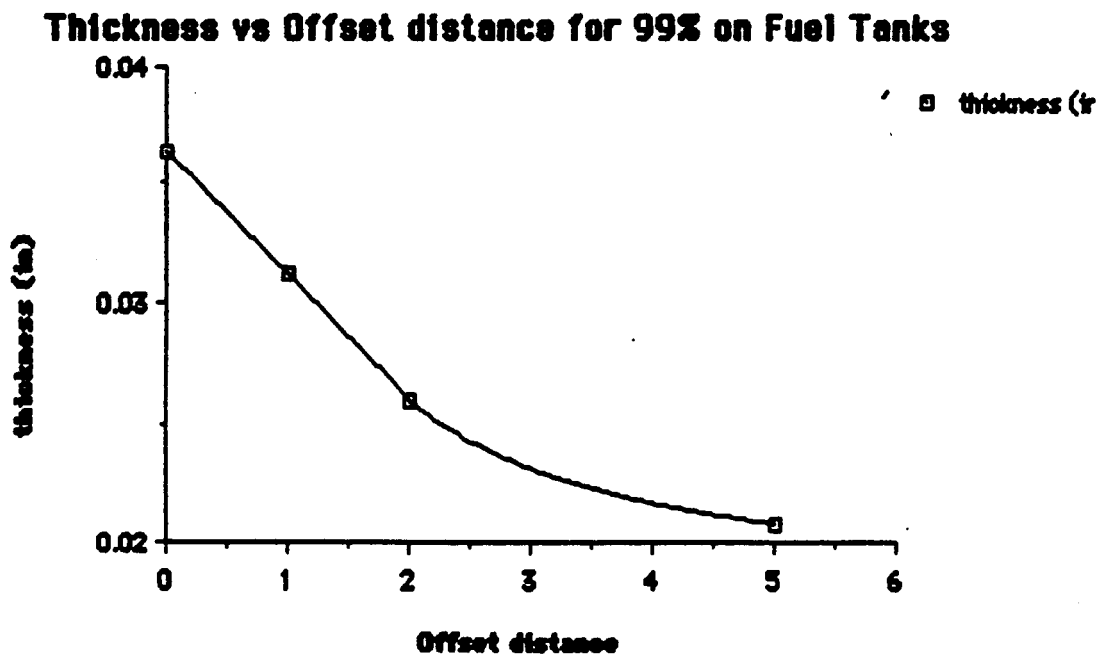
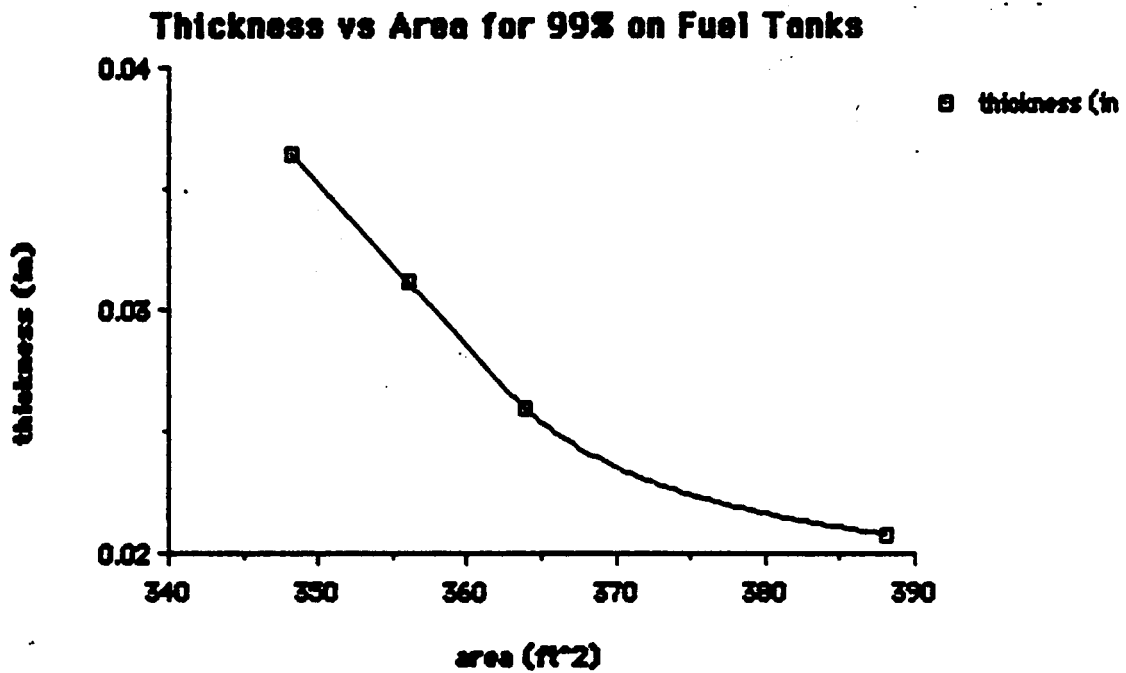
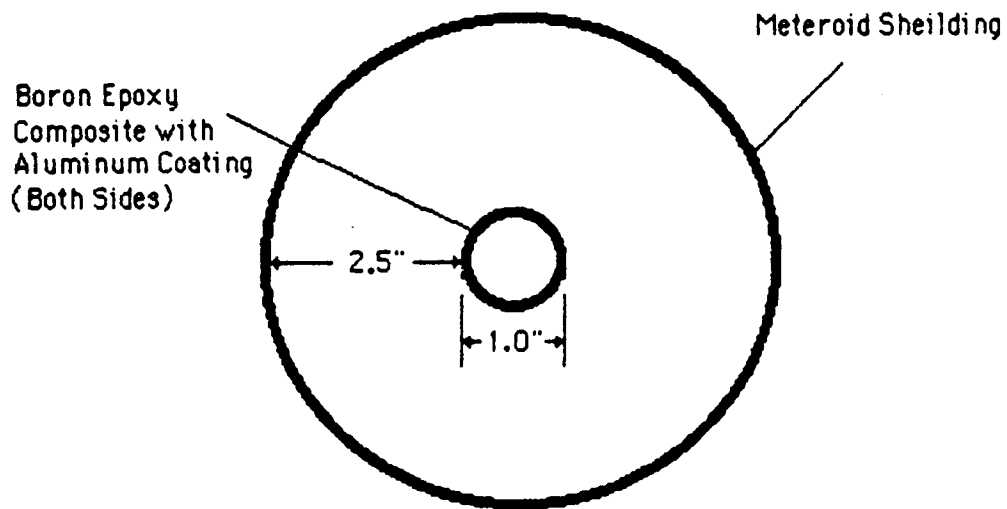
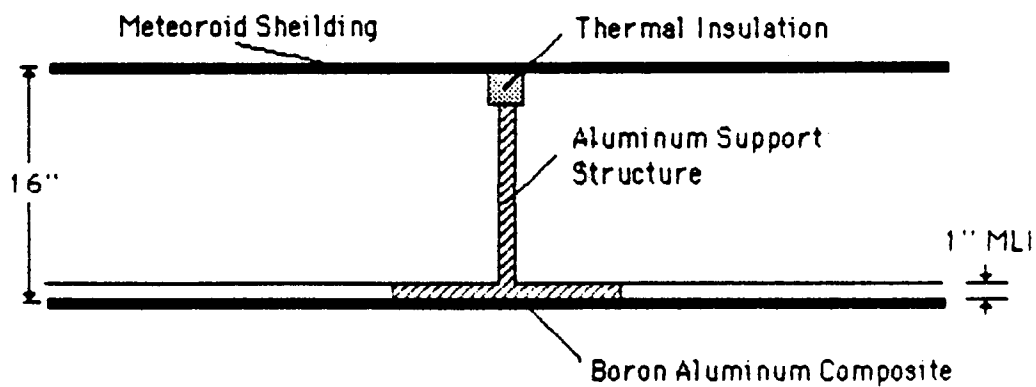


Figure13 - Thickness Us. Offset for Prop. Tank Meteoroid Sheild

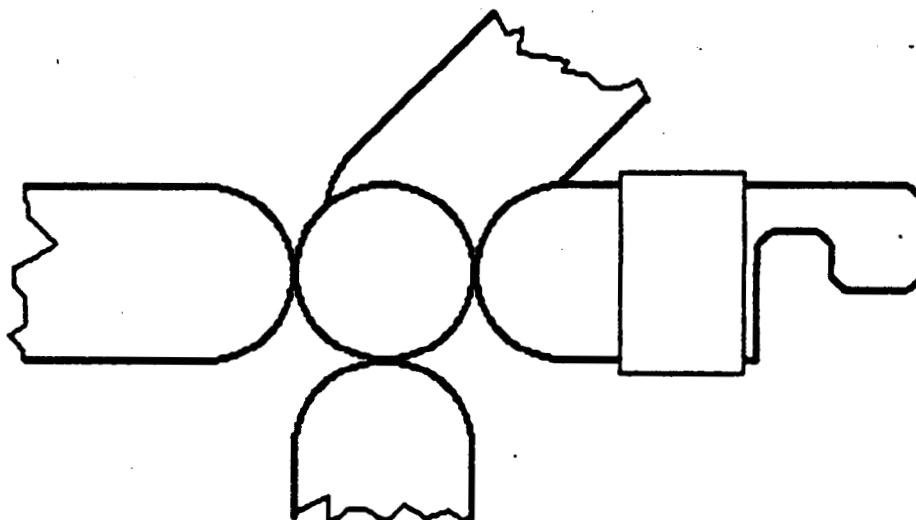


CROSS SECTION OF CARGO BAY TUBING
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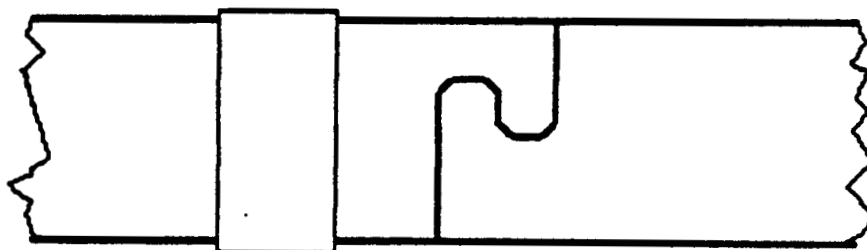


CROSS SECTION OF CABIN WALL STRUCTURE
(NOT TO SCALE)

Figure 14 - USTAR Construction Elements



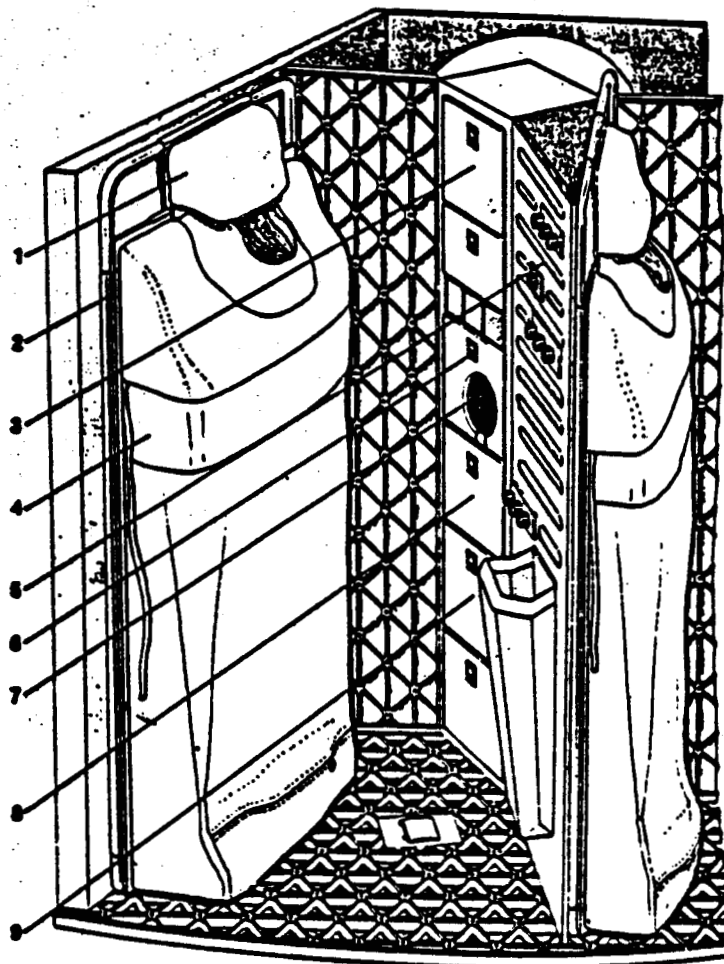
Joint Connector



In-Line Connector

Figure 15 - YSTAR CARGO BAY JOINTS

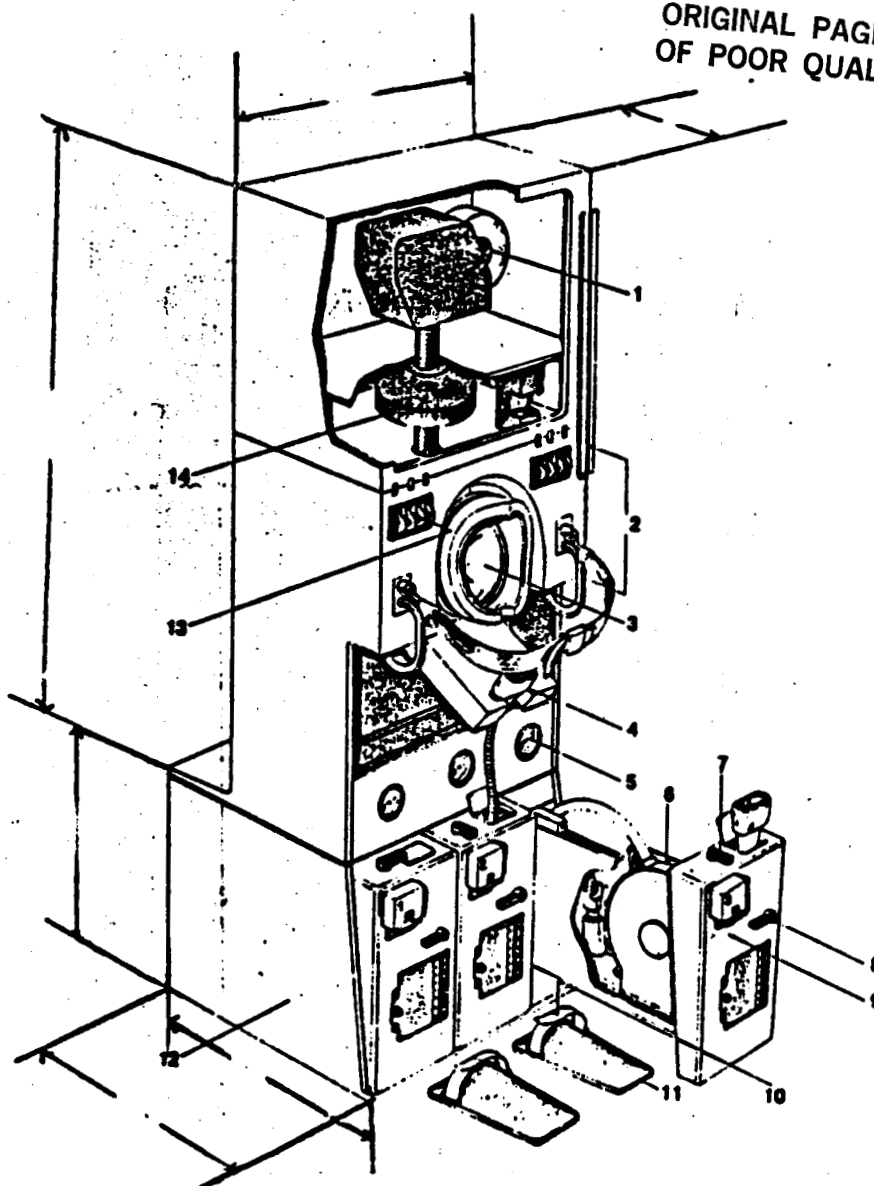
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1. Stowage position for sleep restraint
2. Sleep restraint frame
3. Crew preference kit
4. Sleep restraint
5. Privacy partition
6. Triangle shoes (to fix to floor, grid)
7. Trash container
8. Trash bags
9. Clothing storage

Figure 16 - Vertical Sleeper (Ref.15)

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- | | |
|--|-------------------------------|
| 1. Blower unit | 13. Blower/separator switches |
| 2. Crewman restraints | 14. Fecal collector filter |
| 3. Fecal collector | |
| 4. Urine receptacle (2 positions) | |
| 5. Volume indicator | |
| 6. Urine separator | |
| 7. Airflow valve | |
| 8. Drawer lock/unlock catch | |
| 9. Urine drawer (one for each crewman) | |
| 10. Adjustable velcro attaches | |
| 11. Foot restraints | |
| 12. Fecal/urine collector | |

Figure 17 - Waste Collector (Ref. 27)

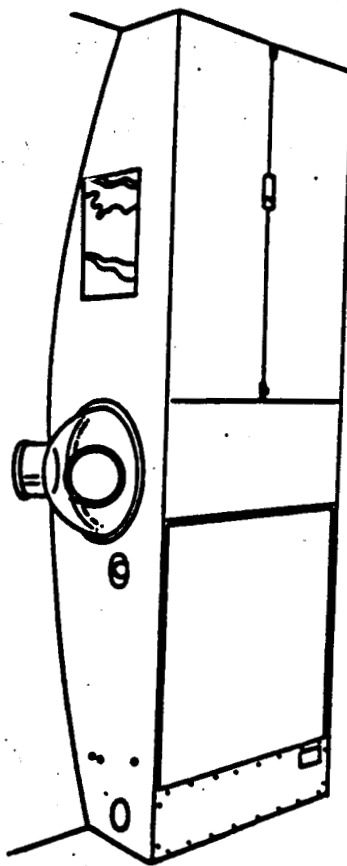


Figure 18 - Food Gallery (Ref. 27)

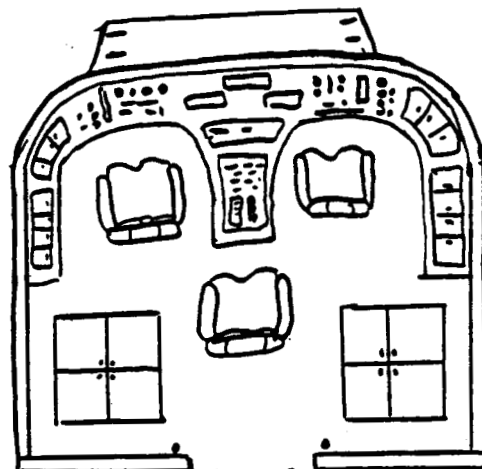
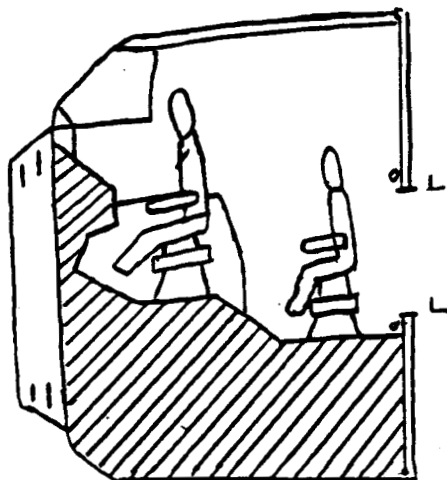
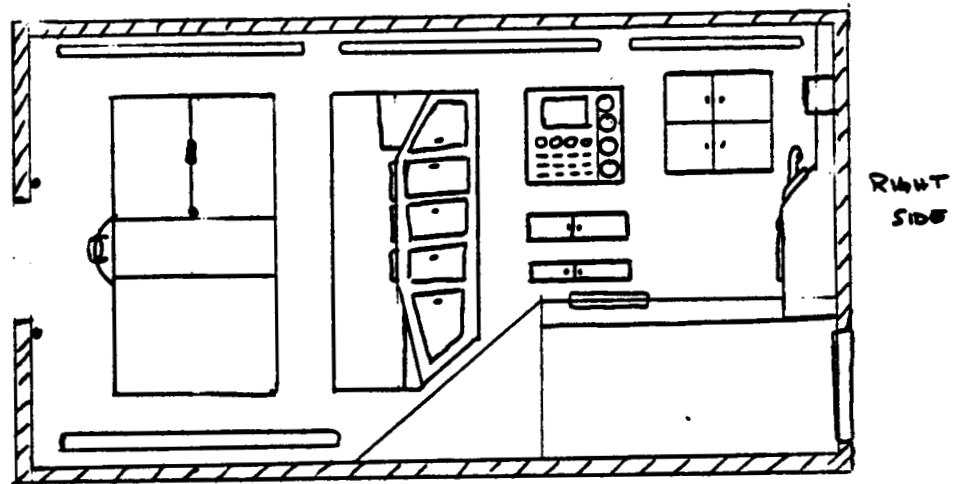
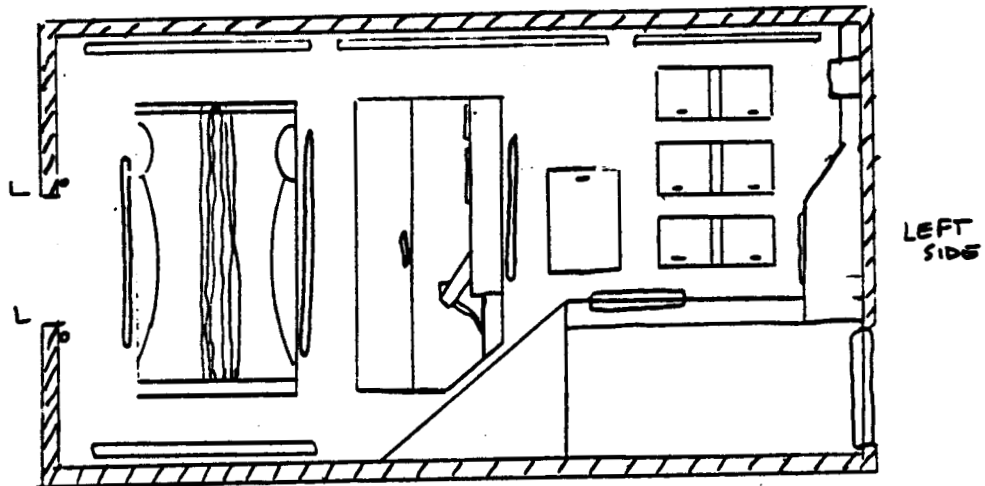


Figure 19 - USTAR Interior Design

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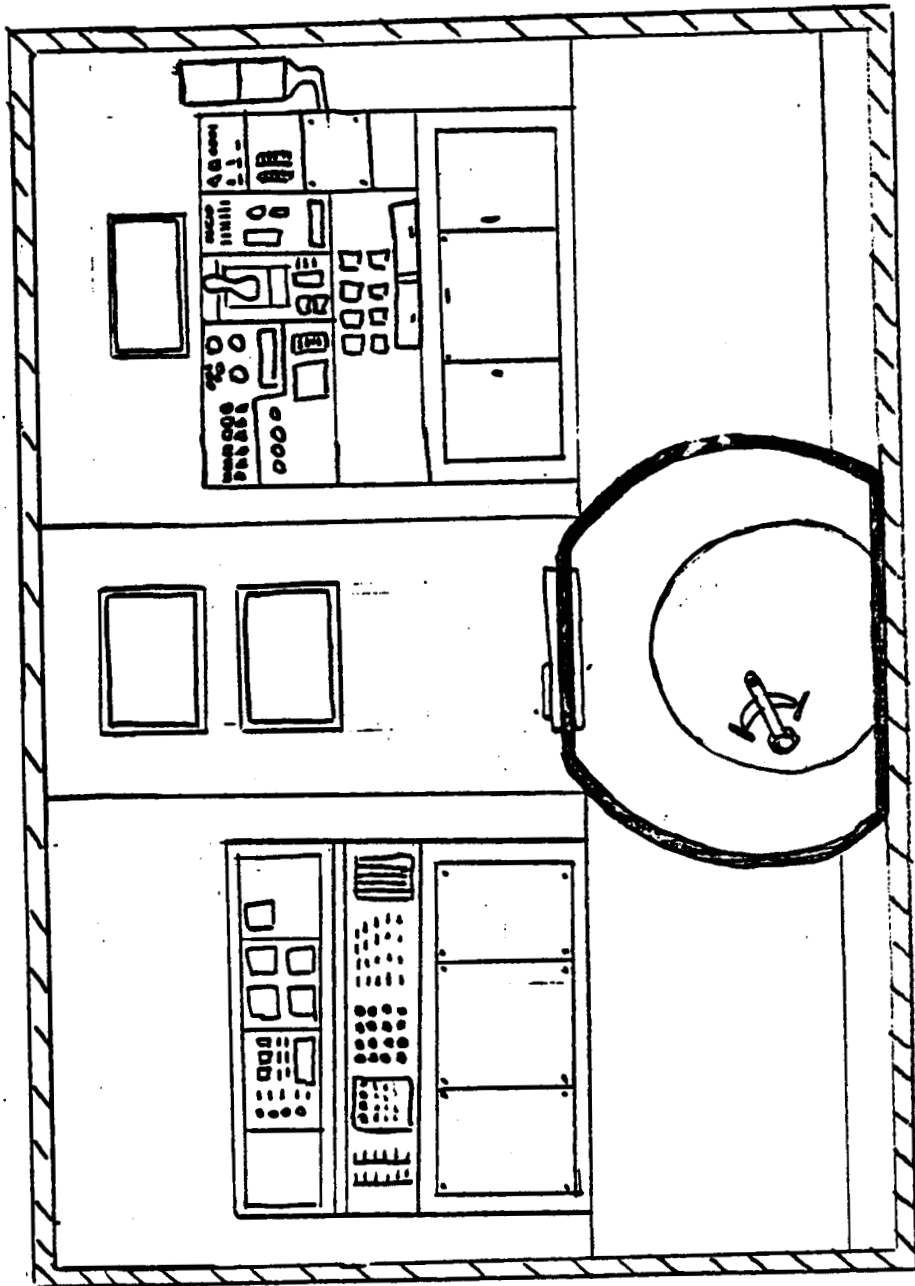


Figure 20 - RMS and TIDS Control Stations

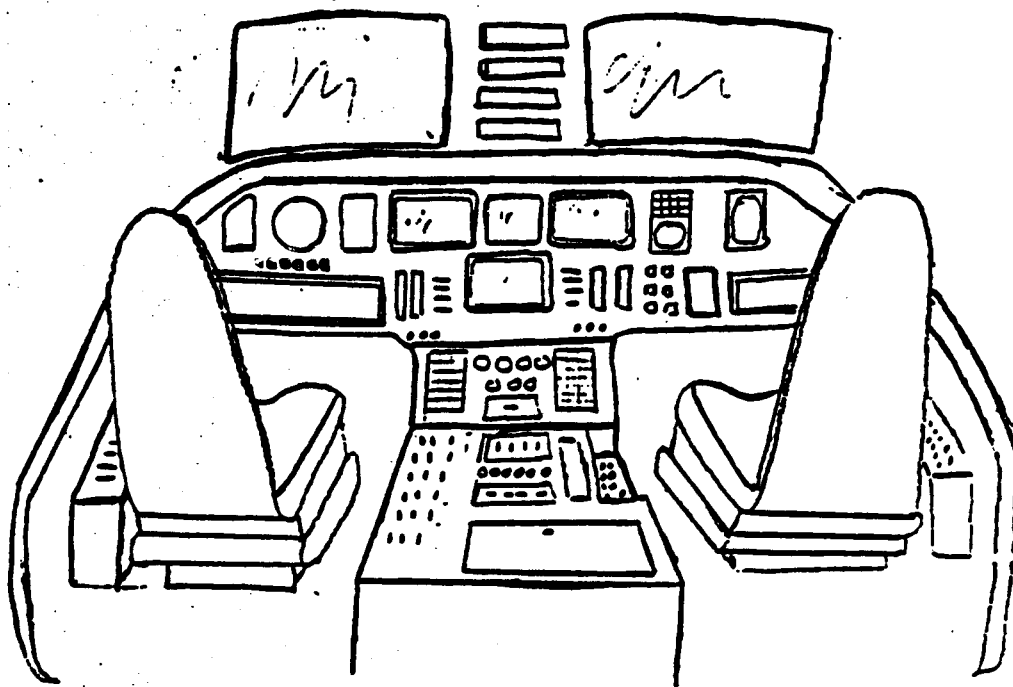


Figure 21 - USTAR Command Deck (Ref.27)

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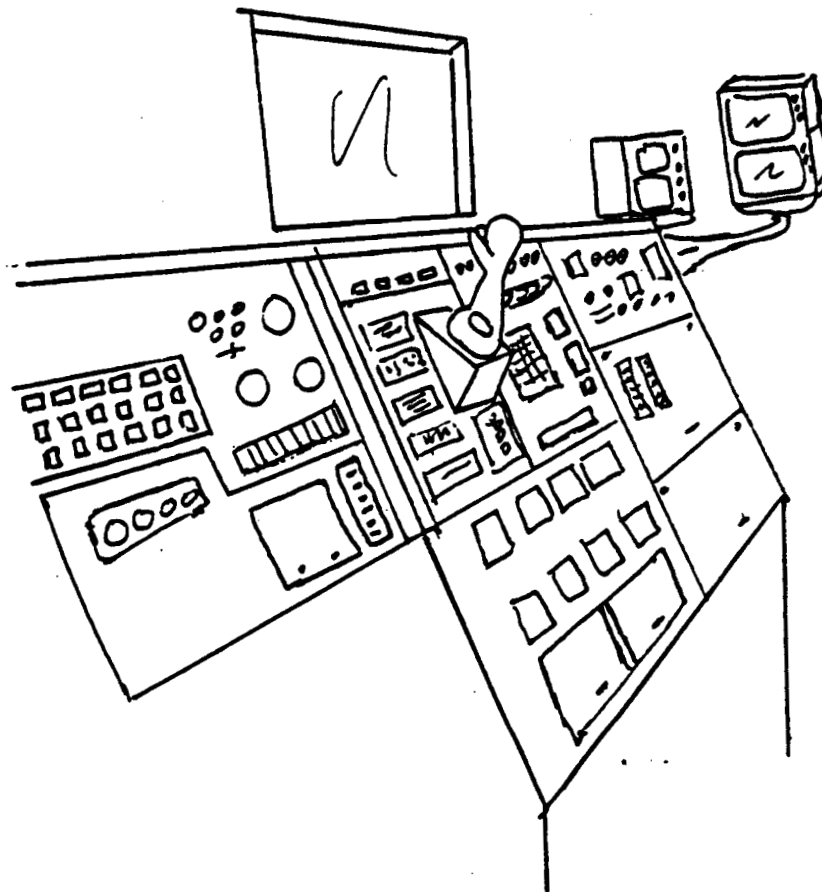


Figure 22 - RMS Control Station (Ref. 27)

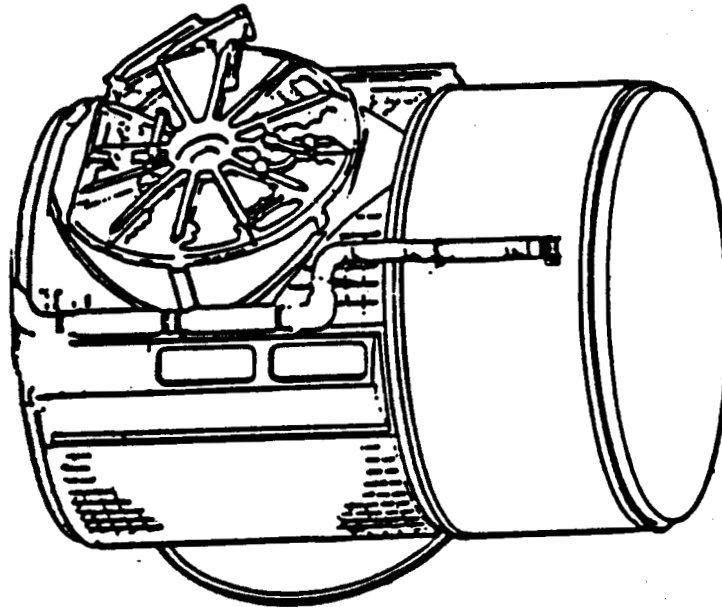
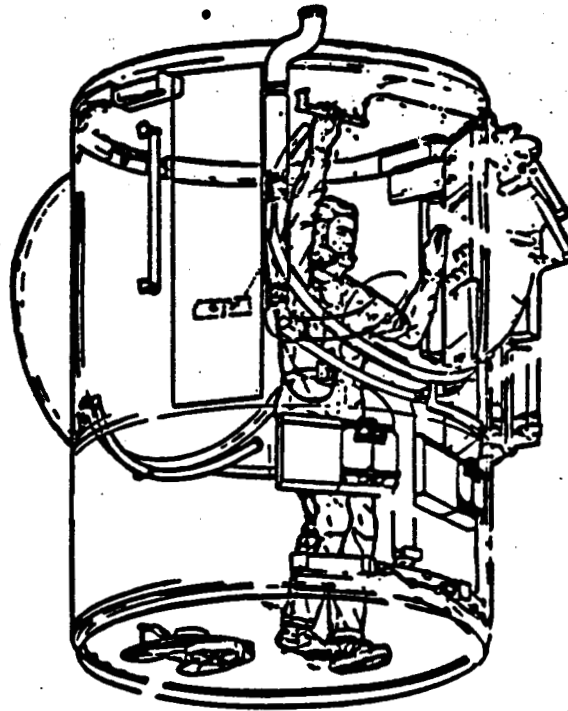
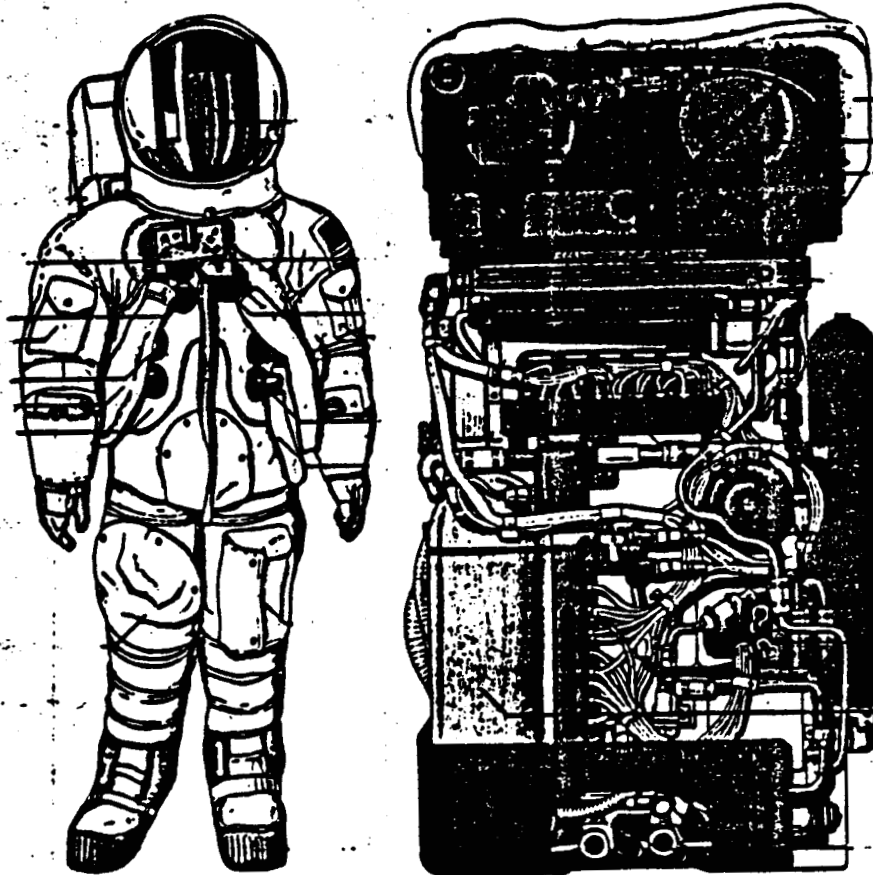


Figure 23 - Airlock for EVA (Ref. 27)

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EMU

PLSS

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Figure 24 - EMU with PLSS (Ref. 27)

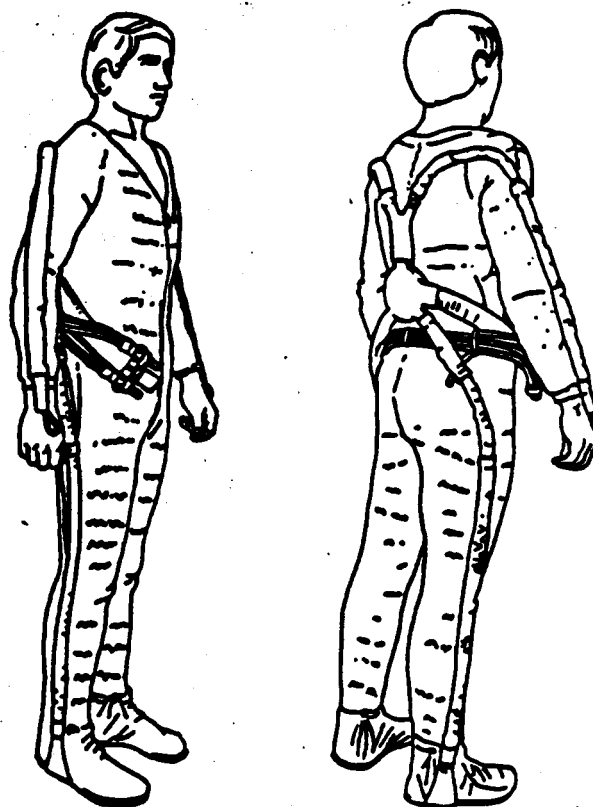


Figure 25 EMU Undergarment (Ref.27)

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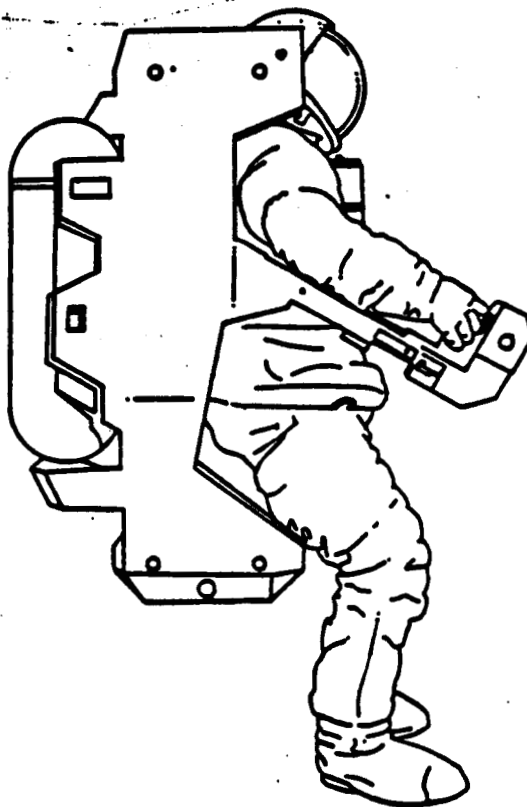
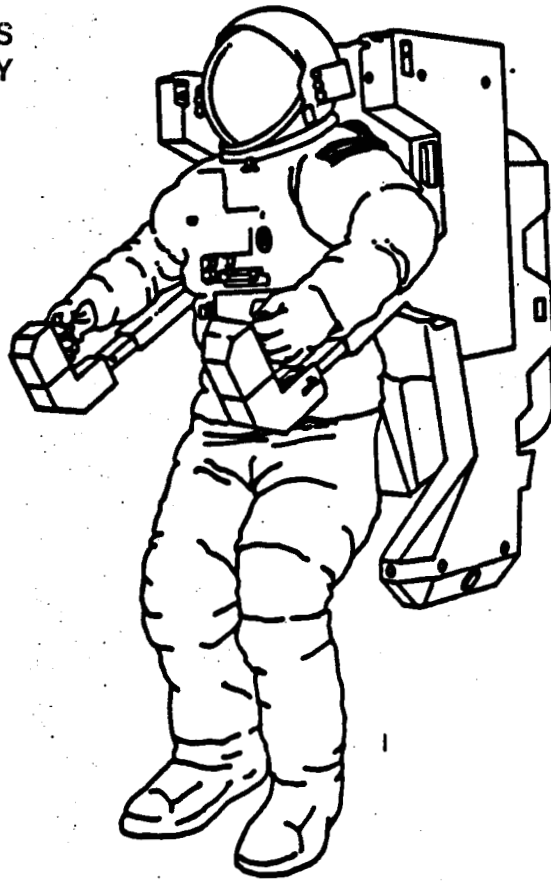


Figure 26 - EMU with MMU (Ref 2.7)

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Figure 27 - USTAR Communication System (Ref.27)

POWER SYSTEM DOMAINS

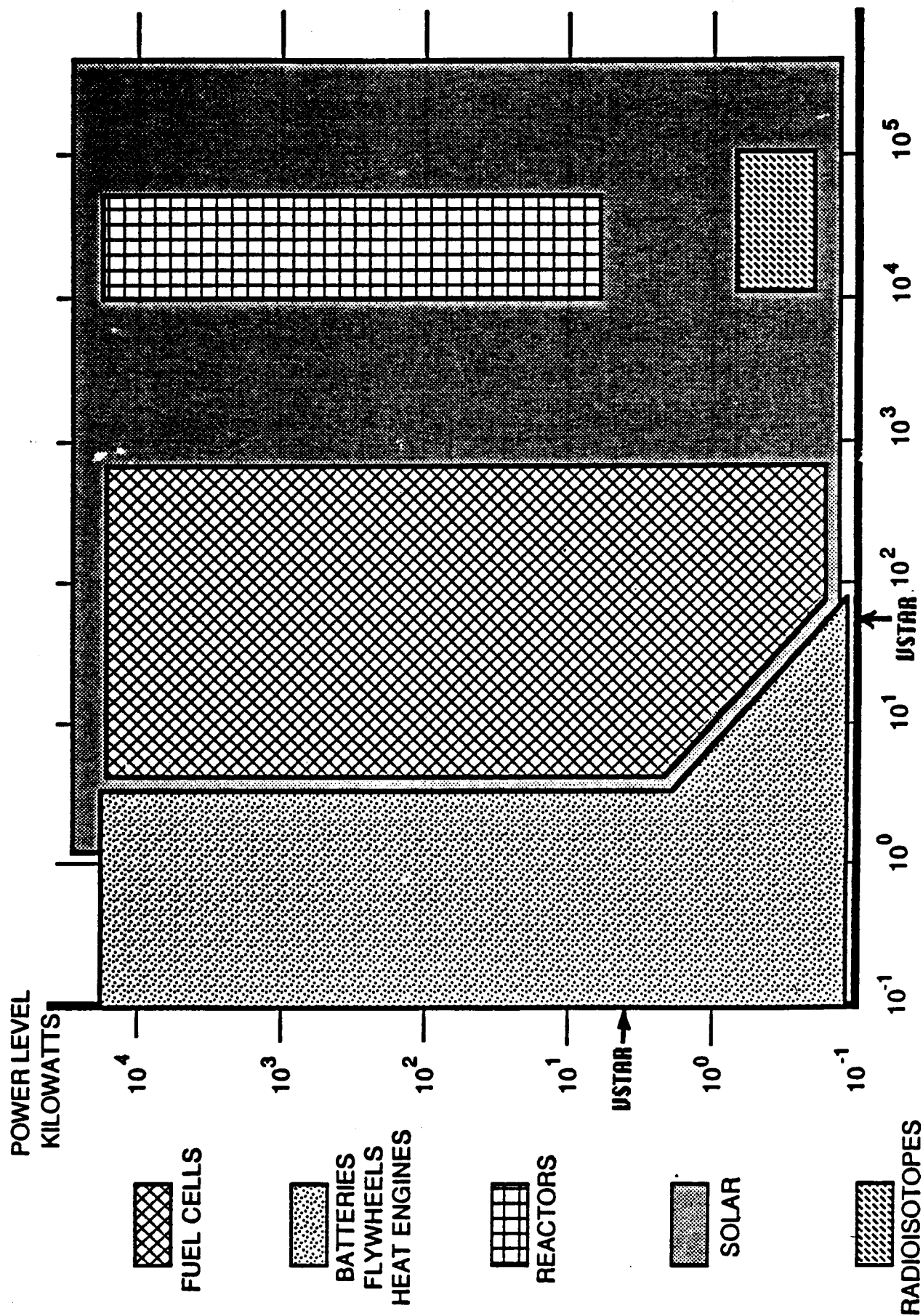
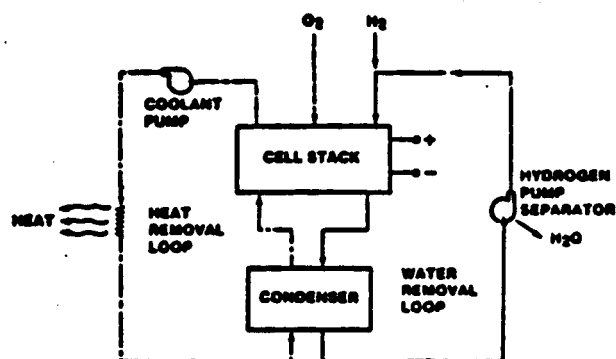
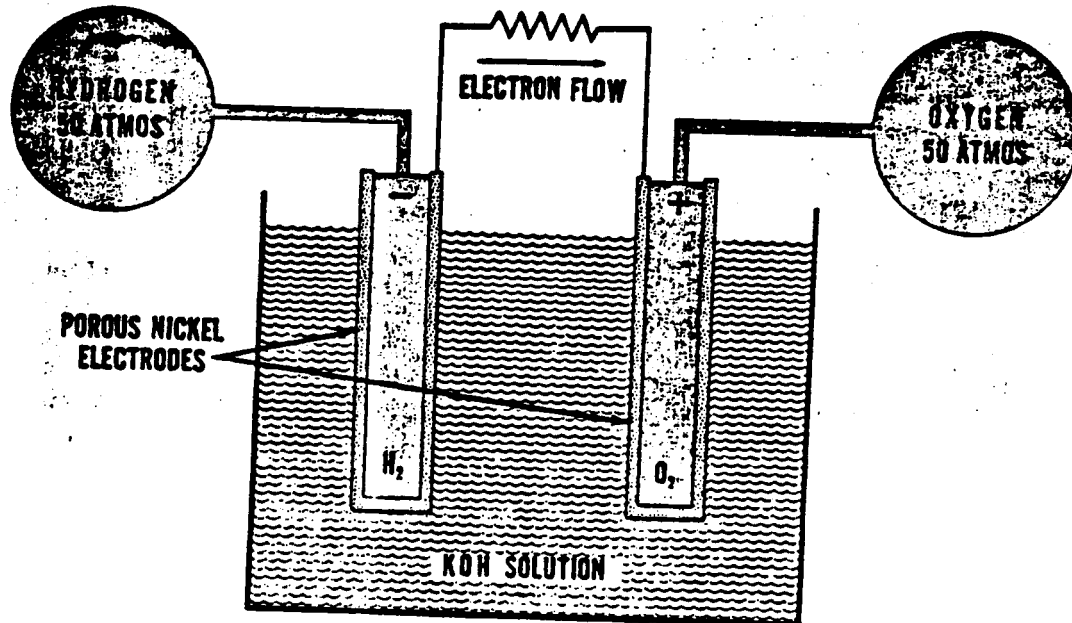


Figure 28 - Power System Domains

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Fuel Cell Circuit



Primary Components

Figure 29 - H_2O_2 Fuel Cell Schematic Diagrams (Ref 23)

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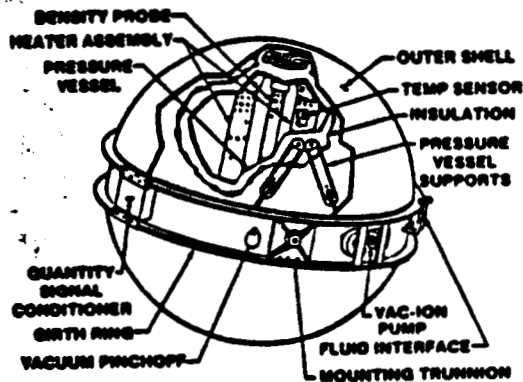


Figure 30 - Oxygen Power Reactant Storage Assembly (PRSA) (Ref. 23)

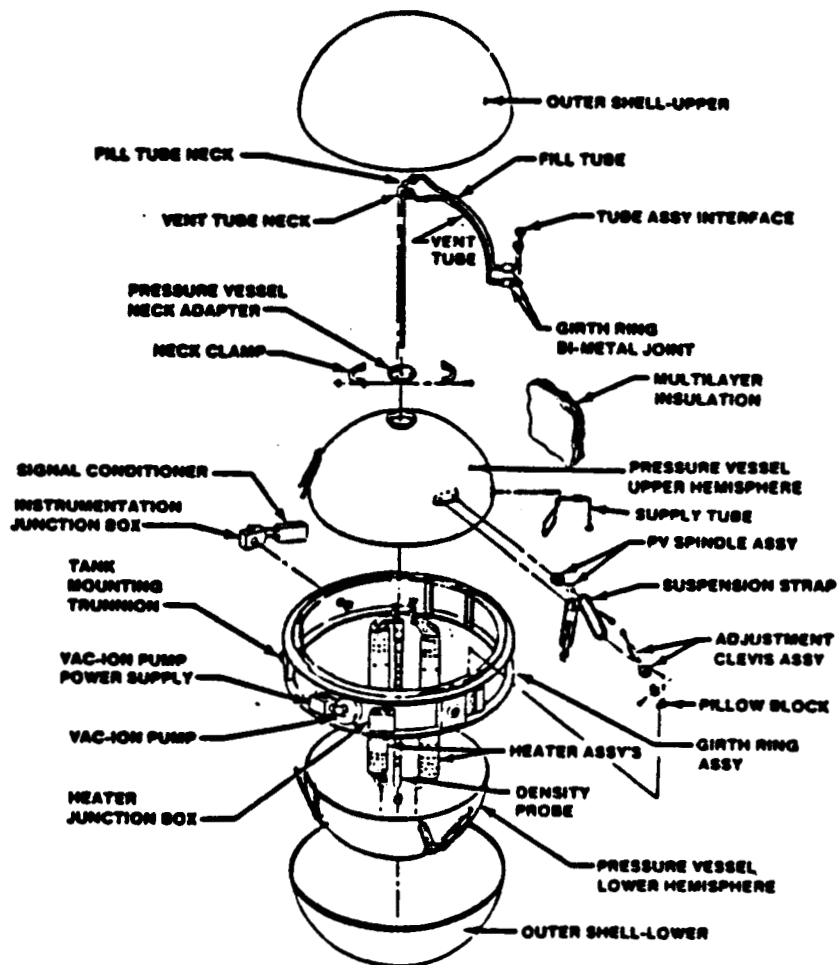


Figure 31 - PRSA Oxygen Tank Configuration (Ref. 23)

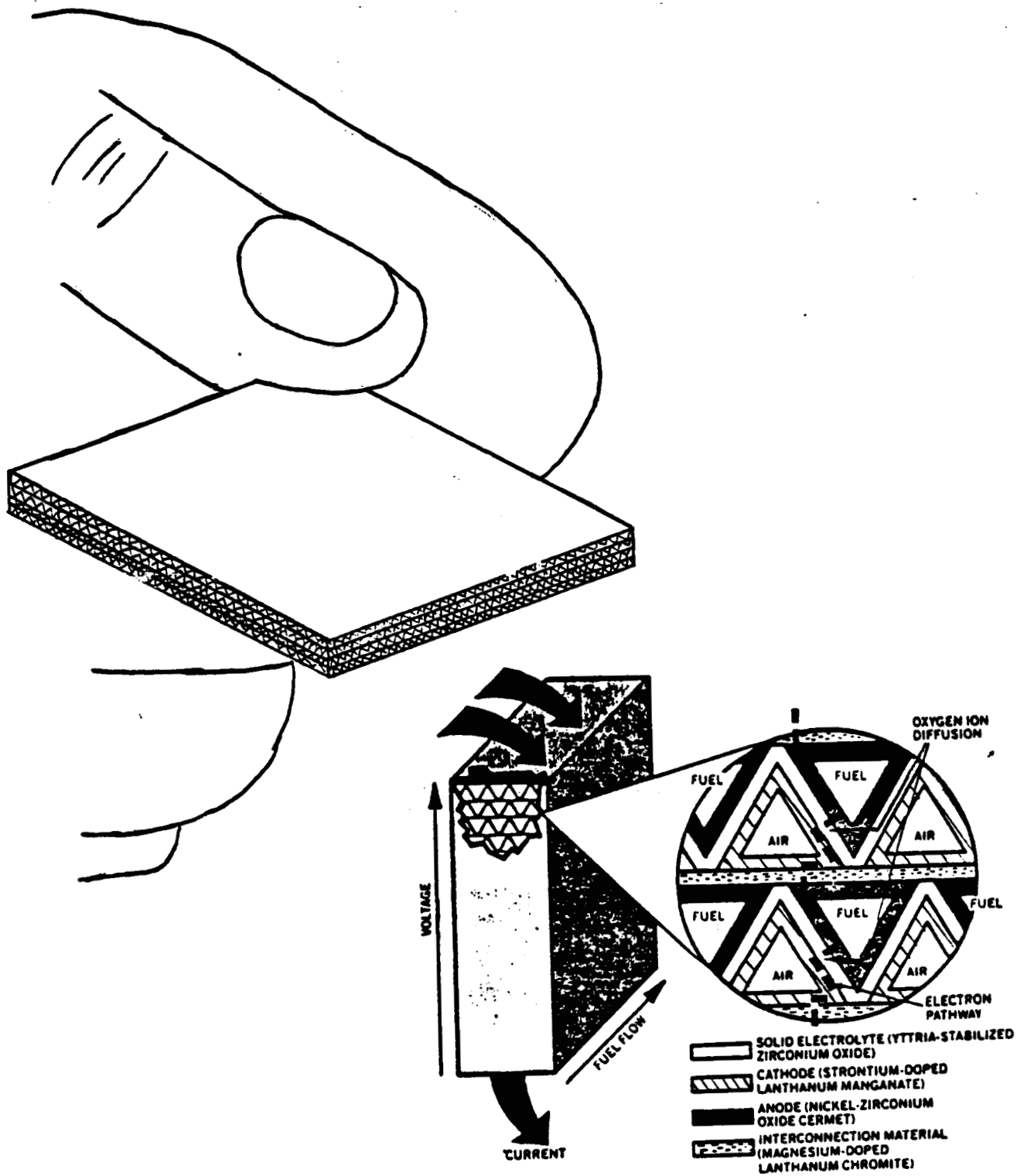
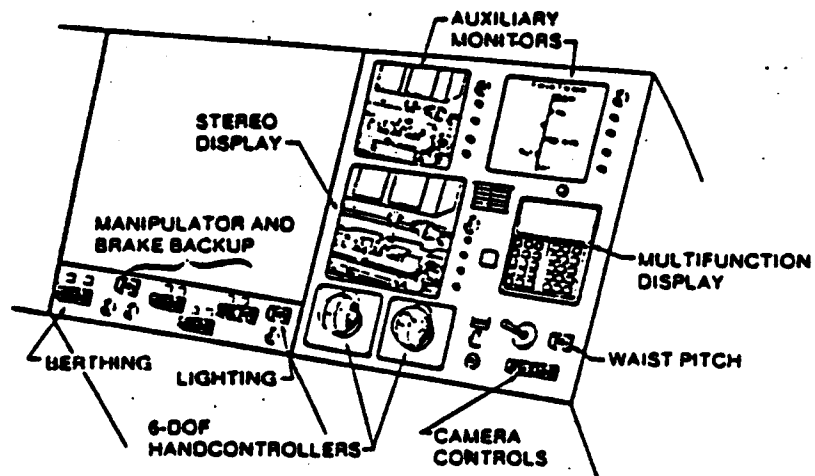


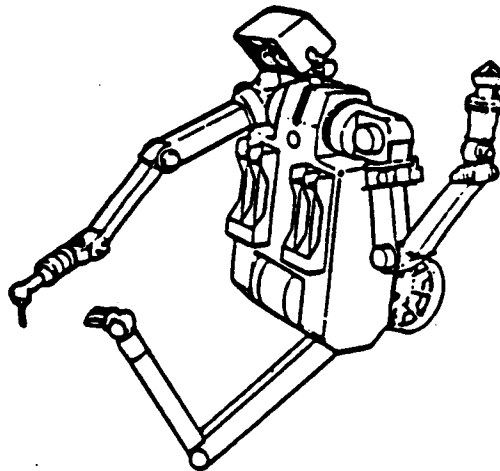
Figure 32 - Multiple-Fuel Fuel Cell (MFFC) (Ref. 3)

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Martin TWS Control Station Concept



Martin Work Station Concept

Figure 33 - Martin TWS Work Station (Ref. 8)

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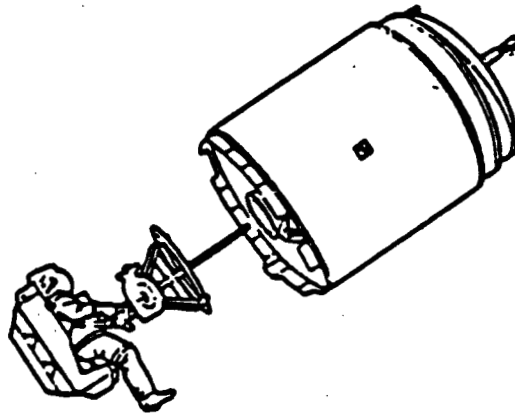


Figure 34.- Approach to Spacecraft with MMU and Stinger (Ref. 8)

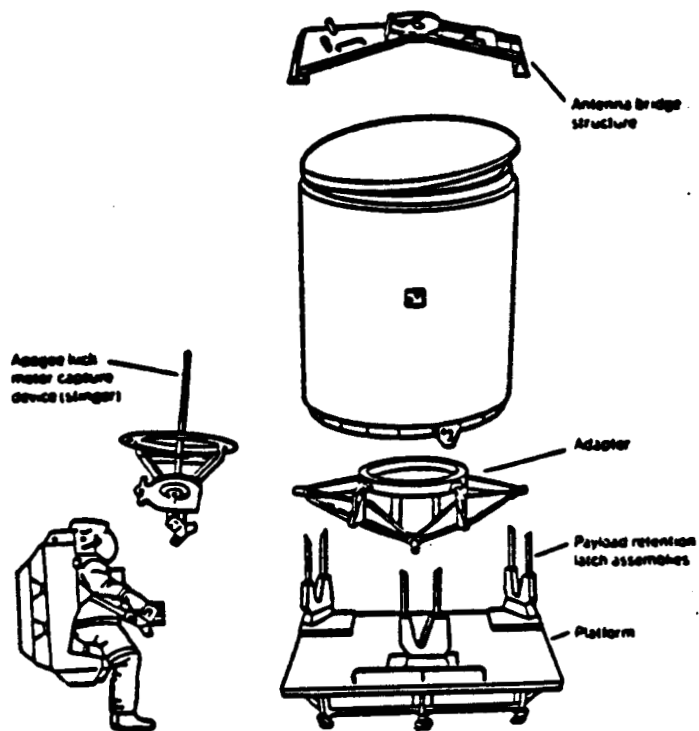


Figure 35.- Spacecraft Recovery Hardware (Ref. 8)

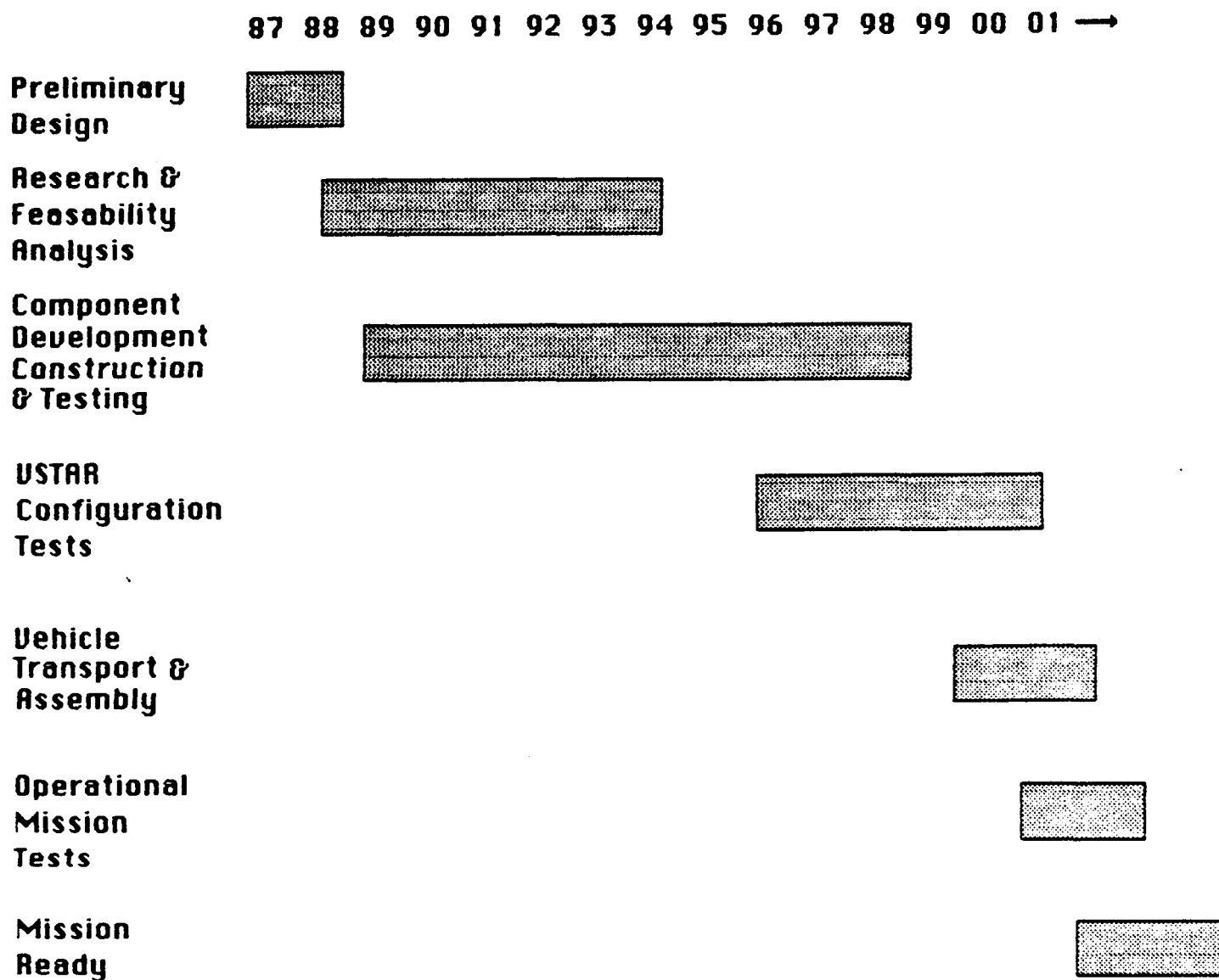


Figure 36 - USTAR Development Timeline

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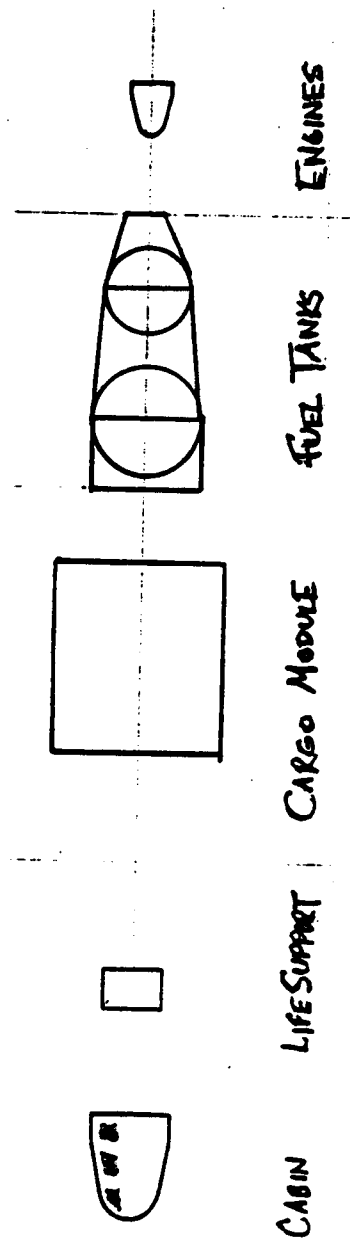


Figure 37 - Idealized Components for Analysis Purposes

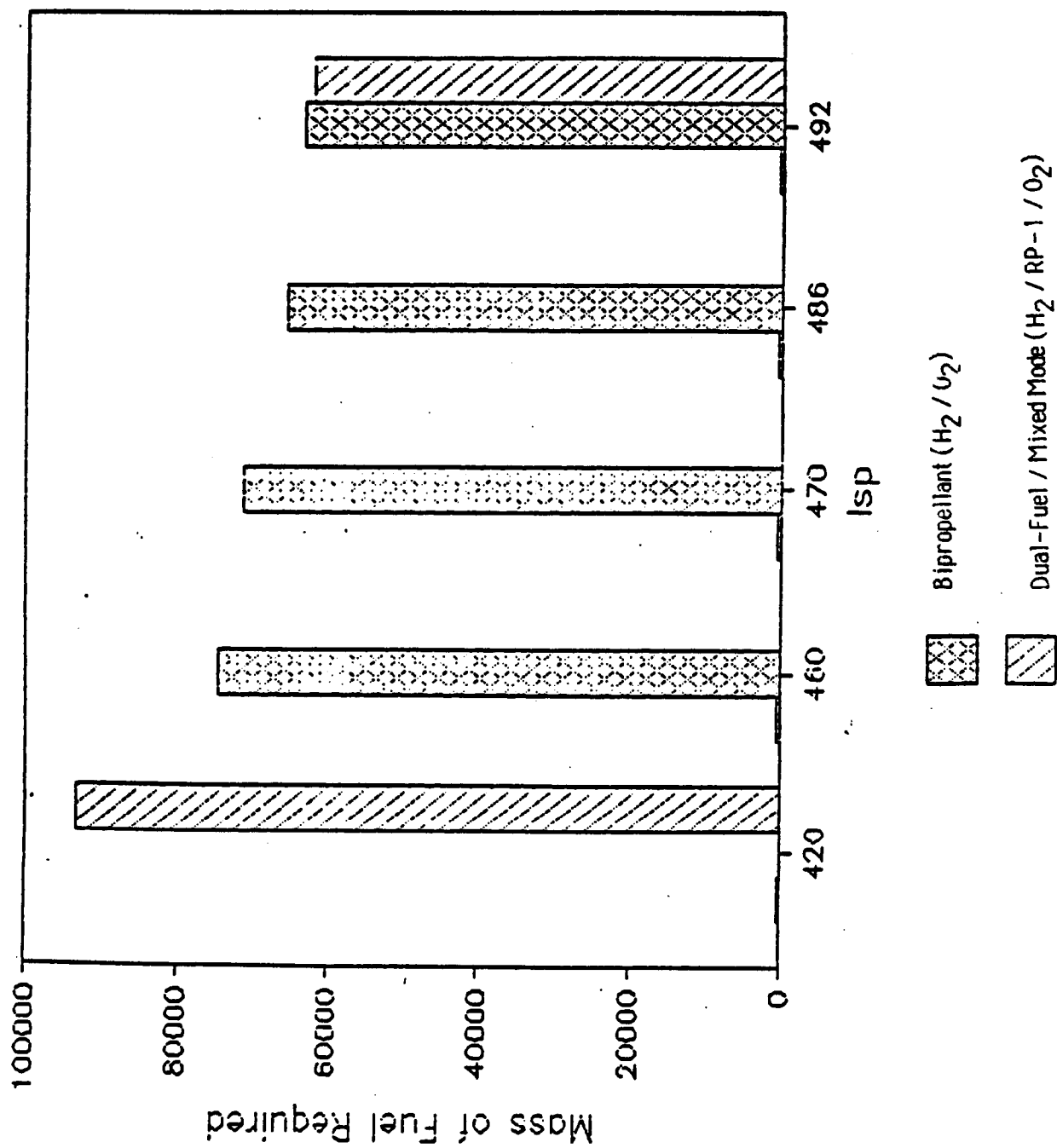


Figure 38 - Fuel Requirements vs. Isp for Bipropellant and Dual-Fuel / Mixed Mode (Tripropellant) Engines

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